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**THREE-AXIS FLUIDIC/ELECTRONIC AUTOMATIC  
FLIGHT CONTROL SYSTEM FLIGHT TEST REPORT**

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**United Aircraft Corporation**

**Prepared for:**

**Army Air Mobility Research and  
Development Laboratory**

**August 1974**

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)  This report covers the flight test of a three-axis Hydrofluidic Stability Augmentation System (HYSAS) coupled with a completely independent parallel attitude and heading hold for a CH-54B helicopter. The design goal was to evaluate the performance of the CH-54B Automatic Flight Control System (AFCS) with the electronic SAS replaced with a fluidic SAS and to establish the technical base for a fluidic inner loop and electronic outer loop control system.		

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ABSTRACT (Continued)

The system was installed in a CH-54B helicopter with its own hydraulic pump and heat exchanger. The dual input servo valves on the AFCS servo were modified by replacing the electrical input transducer with a fluidic input transducer. The motions of the aircraft were recorded by a direct writing recorder. The performance of the helicopter was determined by evaluating the response of the aircraft to pulse inputs. Data was recorded and the system evaluated under the following loading conditions:

- |                |           |
|----------------|-----------|
| (1) CG 336 in. | 35,000 lb |
| (2) CG 346 in. | 47,000 lb |
| (3) CG 335 in. | 28,200 lb |

The aircraft demonstrated handling qualities performance similar to that of the production aircraft. The pitch HYSAS gains were limited by an instability caused by the first vertical bending mode. The yaw gains were limited by noise.

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## PREFACE

This document is the final report on a flight test program authorized by the Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory under Contract DAAG39-73-0237. The technical monitor of this program is Mr. R. P. Smith.

This program is part of the U.S. Army's continuing effort to develop Automatic Flight Control Systems (AFCS) for VTOL aircraft. The objective was to flight test a CH-54B helicopter with a Hydrofluidic SAS and an electronic attitude and heading hold. The work presented started June 25, 1973, and was completed January 25, 1974.

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## INTRODUCTION

This report presents the results of flight testing a three-axis Hydrofluidic Stability Augmentation System (HYSAS) with an independent electronic attitude and heading hold using a CH-54B helicopter as the test vehicle. The primary objective of the flight test program was to demonstrate the feasibility and provide a technical base for a fluidic inner loop and electronic outer loop control system.

The HYSAS was developed under Contracts DAAJ02-68-C-0039 and DAAJ02-69-C-0036 and was flight tested on a UH-1 under Contract DAAJ02-70-C-0017. This system was modified for use in the CH-54B by deleting the yaw axis pilot input loop and by adding a new shaping network for the pitch axis. The AFCS servo was modified to accept fluidic signals in lieu of electrical signals, making the integration of the HYSAS quite simple and the possibility of a retrofit attractive.

The AFCS was evaluated analytically on both the Sikorsky hybrid simulator and the CH-54B helicopter. The evaluation consisted of determining the response, damping and ability to return to trim following pulse inputs. Pilot comments were evaluated for maneuvering flight.

The simulation results are recorded in Appendix I. The AFCS flight test results are recorded in Appendix II, and the pilot evaluations are contained in Appendix III.

### SCOPE OF PROGRAM

The present production CH-54B is equipped with a dual electronic stability augmentation system (SAS) and an outer loop attitude-heading hold system. The electronic SAS was deactivated and was replaced with a Hydrofluidic Stability Augmentation System (HYSAS). The program consisted of two major tasks: (1) Design, Fabrication and Installation and (2) Test and Evaluation.

The design, fabrication and installation (Task 1) consisted of the following items:

- (a) The CH-54B with the HYSAS was simulated using a six degree of freedom simulation.
- (b) The electric torque motor on the servo valves was replaced with a fluidic flapper assembly to provide the required interface between the fluidic sensor/control assembly and the CH-54B AFCS servo.
- (c) The three-axis fluidic SAS that was developed under Contract DAAJ 02-70-C-0017 was modified for use on the CH-54B.
- (d) The three-axis fluidic SAS was integrated with the modified AFCS servo system, and integration tests were performed in the laboratory.
- (e) The CH-54B AFCS components were modified and the fluidic system was installed and ground checked.

The test and evaluation phase consisted of the in-flight evaluation of the HYSAS coupled with the attitude-heading hold system. Listed below are the combinations of gross weight and center of gravity that were flown:

<u>Loading Conditions</u>	<u>Gross Weight (lb)</u>	<u>CG Locations (in.)</u>
1	35,000	336
2	47,000	346
3	28,200	335

## DESCRIPTION OF SYSTEM

The Automatic Flight Control System (AFCS) tested is composed of three channels: pitch, roll, and yaw. Each AFCS channel consists of a hydrofluidic inner loop SAS and an independent electronic outer loop attitude or heading hold system.

### THE PITCH CHANNEL

Figure 1 shows a block diagram of the pitch AFCS channel. The HYSAS provides the basic airframe damping and stability augmentation. The pitching rate is detected by the pitch vortex rate sensor. This signal is amplified and shaped to provide rate plus lagged rate compensation, which results in good long- and short-term damping. The signal is further shaped to provide a "washout" or a high pass to eliminate all steady-state pitch rates encountered in coordinated turns. The signal is then introduced into the pitch servo valve where it is summed in series (inner loop) with the pilot's controls.

The attitude control circuitry accepts the gyro displacement signal directly. The attitude signal is passed through an automatic synchronizing circuit which is activated whenever the trim release or beeper trim is actuated. In the synchronizing mode, the pilot can retrim the aircraft attitude by simply flying to the new attitude and engaging trim. While establishing the new attitude, the synchronizing circuit is engaged ( $S_1$ ) which counters the change in gyro output, resulting in a zero net reference signal. Simultaneously, the position sensor clutch ( $S_2$ ) is released, nulling the position sensor. When the trim is engaged, the synchronizing circuit  $S_1$  opens, the clutch  $S_2$  engages, and a new trim reference and attitude is established. The established attitude is then shaped with rate plus proportional compensation and directed to the outer loop actuator. The outer loop actuator drives the cyclic stick and the position sensor until its output cancels the gyro error signal, at which time the outer loop actuator is halted. The derived rate signal provides the required damping to keep the outer loop stable.

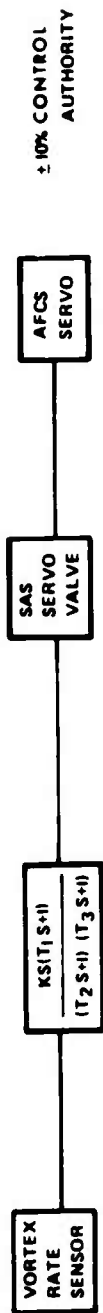
### THE ROLL CHANNEL

Figure 2 shows a block diagram of the roll AFCS channel. The HYSAS provides the basic airframe damping and stability augmentation. The rolling rate is detected by the roll vortex rate sensor. This signal is amplified and shaped to provide a "washout" or high pass which eliminates all steady-state effects of the fluidics. The signal is then introduced into the roll servo valve where it is summed in series with the pilot's controls.

The attitude control circuitry is identical to that described under the pitch channel.

### THE YAW CHANNEL

Figure 3 shows a block diagram of the yaw AFCS channel. The HYSAS provides the basic airframe damping and stability augmentation. The yawing rate is



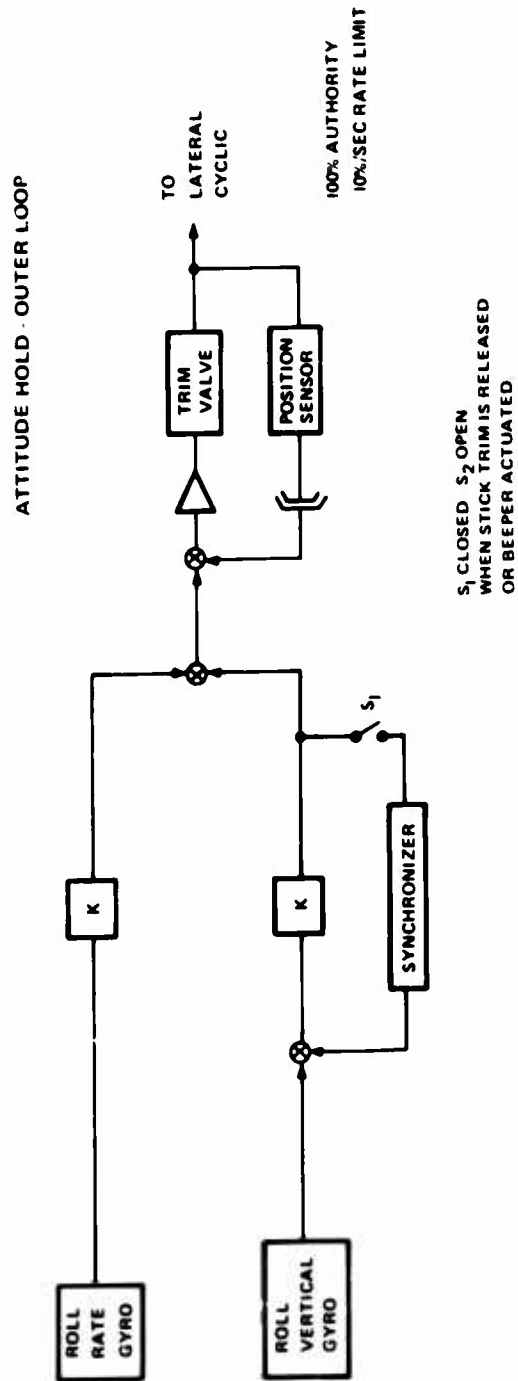
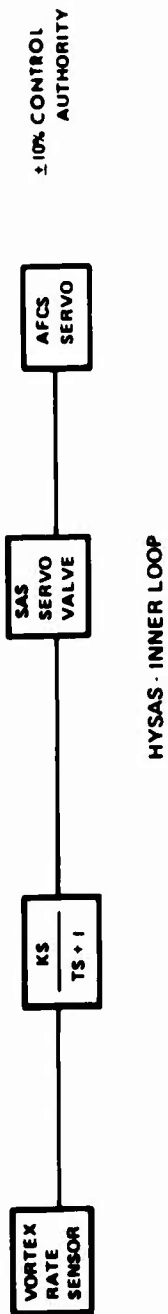


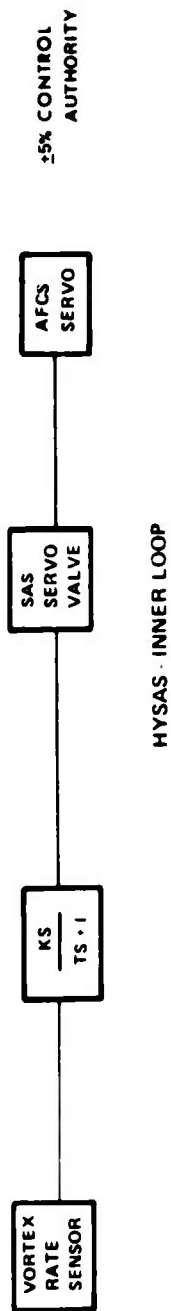
FIGURE 2 CH-54B ROLL AXIS.

detected by the yaw vortex rate sensor. This signal is amplified and shaped to provide a "washout" which eliminates saturation in a steady-state turn. The signal is then introduced into the yaw servo valve where it is summed in series with the pilot's controls.

The heading error signal is generated by the directional gyro portion of the compass system. This signal is synchronized whenever a pedal switch is actuated. In this mode the pilot can retrim the aircraft heading by flying to the desired heading and releasing the pedals. While flying to a new heading, the synchronizing circuit nulls the combination of the directional gyro and the yaw trim, resulting in a zero net reference signal when the desired heading is attained. Actuation of the pedal switch also opens the position sensor clutch ( $S_2$ ), allowing the sensor to return to its null position. Releasing the pedal switch releases the synchronizing circuit ( $S_1$ ), engages the position sensor clutch ( $S_2$ ), and establishes the heading reference at that point.

The heading error signal is directed in two paths, one proportional and the other through an integrator circuit. This significant difference between the yaw heading hold function and that of attitude retention is the integration. In the attitude hold system the control system responds to attitude errors with a control position change proportional to the error. As the aircraft responds to the control correction, the error is reduced and the control correction is reduced. As the error approaches zero, the correction control motion approaches zero. In the yaw system where large steady-state disturbing forces are common in the form of collective changes, a steady-state heading error is developed. The integrator senses the error and continues to drive a signal until the error is returned to zero.

An additional feature of the yaw system is the trim capability in which the heading reference can be altered by adjusting the yaw trim knob.



HEADING HOLD - OUTER LOOP

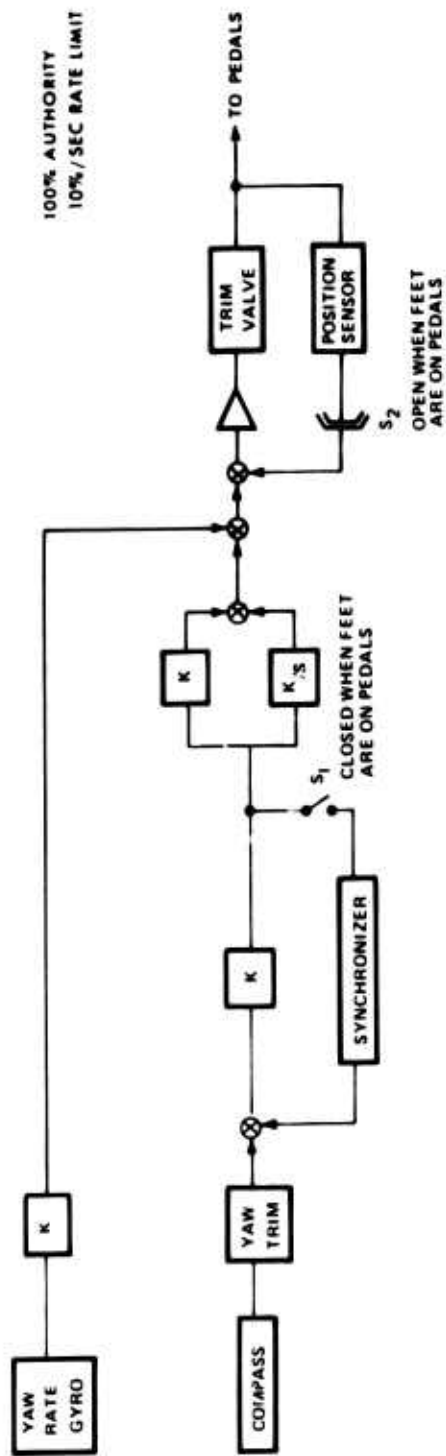


FIGURE 3. CH-54B YAW AXIS.

## DESIGN, FABRICATION AND INSTALLATION

### SIMULATION

The equations of motion for the CH-54B were used to determine the HYSAS and attitude-heading hold design requirements. A dynamic model of the proposed system was used to determine the fluidic transfer functions required to duplicate the performance of the production CH-54B inner loop electronic SAS.

The simulation was performed on the Sikorsky hybrid computer complex, which consists of the following major components:

1. Two Applied Dynamics model AD/4 analog computers.
2. One Digital Equipment Corporation model PDP-10 digital computer.
3. One hybrid interface unit.
4. Various input/output peripherals.

The real-time CH-54B simulation had the following capabilities:

1. Six fuselage degrees of freedom
2. Nonlinear fuselage aerodynamic data.
3. Six blades, each with feathering and flapping degrees of freedom.
4. Nonlinear rotor blade airfoil section aerodynamic data with stall and Mach number effects.

TABLE I. CH-54B HYSAS SIMULATION TEST ENVELOPE

	CONDITION 1	CONDITION 2	CONDITION 3
GROSS WEIGHT, lb	42,000	34,000	47,000
CENTER OF GRAVITY, in.	336	328	346
$I_x$ , slugs	43,500	43,500	43,500
$I_y$ , slugs	195,892	195,892	195,892
$I_z$ , slugs	176,000	176,000	176,000

The loading conditions and inertias used in the simulation are depicted in Table 1. The coupled SAS and attitude-heading hold was evaluated at hover, 60 knots and 100 knots. The time histories are contained in Figures 28 to 36.

### Pitch Channel

The pitch channel displays good attitude hold with the damping ranging from 0.7 in hovering flight to 0.5 at the extreme aft CG and 100 knots. The steady-state change in attitude noted in hovering flight is due to translational lift. The cyclic motion that is observed immediately after the pulse disturbance is the attitude hold system reacting to the simulated disturbance. Detailed review of the data indicates that the simulated HYSAS stabilized aircraft duplicates the production CH-54B pitch axis. The required transfer functions are

$$\text{PITCH HYSAS} \quad \frac{B_{1s}}{\dot{\Theta}_f} = 0.91 \left( \frac{2.12 S + 1}{8 S + 1} \right) \left( \frac{10 S}{10 S + 1} \right) \%/\text{deg/sec} \quad (1)$$

$$\text{ATTITUDE HOLD} \quad \frac{B_{1s}}{\Theta_f} = (0.02 S + .85) \%/\text{deg} \quad (2)$$

### Roll Channel

The roll channel displays excellent attitude hold with the system damping being measured at critical for all flight regimes simulated. Most pilots prefer the roll damping to be critical to allow them to transition smoothly from one bank angle to another. The cyclic motion is due to the roll attitude hold system reacting to the disturbance. A detailed review of the roll data indicates that the simulated HYSAS stabilized aircraft duplicated the production CH-54B. The required transfer functions were found to be

$$\text{ROLL HYSAS} \quad \frac{A_{1s}}{\phi} = 0.16 \left( \frac{10 S}{10 S + 1} \right) \%/\text{deg/sec} \quad (3)$$

$$\text{ATTITUDE HOLD} \quad \frac{A_{1s}}{\phi} = (0.13 S + 1.25) \%/\text{deg} \quad (4)$$

### Yaw Channel

The yaw channel displayed good heading hold response to the simulated disturbance. The damping was observed to be 0.5 for all regimes of flight. The system performance in a hover at the three simulated weights was noted to be slightly inferior to the production CH-54B. The difference noted here is due to the high pass filter required in the yaw HYSAS. The production system does not use a high pass filter under 60 knots. Performance of the system in forward flight duplicated the production CH-54B. The gains required to achieve the performance noted are

$$\text{YAW HYSAS} \quad \frac{\dot{\Theta}_{TR}}{\dot{\Psi}} = 0.525 \left( \frac{2S}{2S+1} \right) \%/\text{deg/sec} \quad (5)$$

$$\text{HEADING HOLD} \quad \frac{\dot{\Theta}_{TR}}{\dot{\Psi}} = \left( 0.705S + 5.0 + \frac{0.1}{S} \right) \%/\text{deg/sec} \quad (6)$$

## DESIGN AND FABRICATION

### Sensor Controllers

The design and fabrication consisted of converting the three-axis HYSAS, previously developed under Contract DAAJ02-70-C-0017 for the UH-1 helicopter, for use on the CH-54B helicopter.

#### Pitch Axis

The pitch sensor/controller consists of a vortex rate sensor, four hydrofluidic amplifiers, a lead lag and a high pass (washout) network. A null adjustment is located on the unit to allow initial system set-up. A built-in test button is also located on the unit to provide a degree of self test. The high pass filter time constant was increased to 10 seconds. The gain and the lag time constant of the lead-lag network was also increased. Figure 4 is a schematic of the pitch axis.

#### Yaw Axis

The yaw sensor/controller consists of a vortex rate sensor, three hydrofluidic amplifiers, and a high pass filter. A null adjustment button and a built-in test button are located on the unit similar to the pitch axis. An increase in gain was required for the CH-54B installation and was accomplished with adapter blocks and a high gain amplifier. The pilot pedal feed forward loop required for the UH-1C helicopter was removed. Figure 5 is a schematic of the yaw axis.

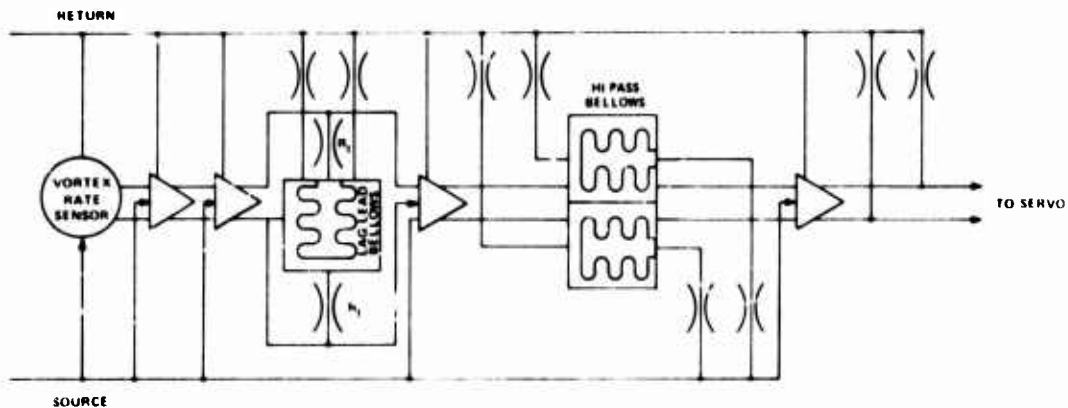
#### Roll Axis

The roll sensor/controller is similar in complexity to the modified yaw controller. It consists of a vortex rate sensor, two hydrofluidic amplifiers, and a high pass filter. The unit also contains a null adjustment and a built-in test button. Figure 6 is a schematic of the roll axis.

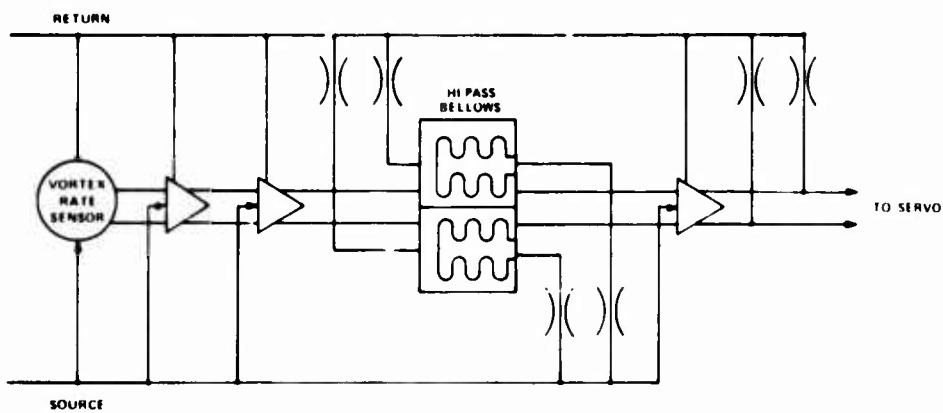
#### AFCS Servo

The AFCS servo provides the pilot with four main functions on the CH-54B:

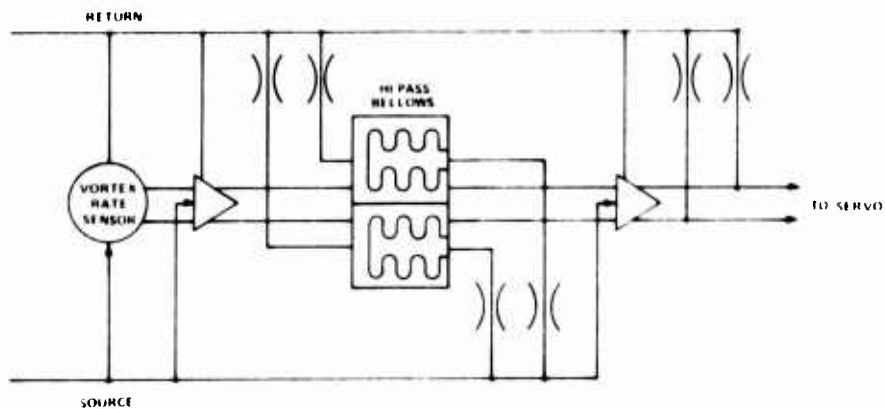
Flight control friction and inertia elimination.



**FIGURE 4. PITCH SENSOR/CONTROLLER.**



**FIGURE 5. YAW SENSOR/CONTROLLER.**



**FIGURE 6. ROLL AXIS SENSOR/CONTROLLER.**

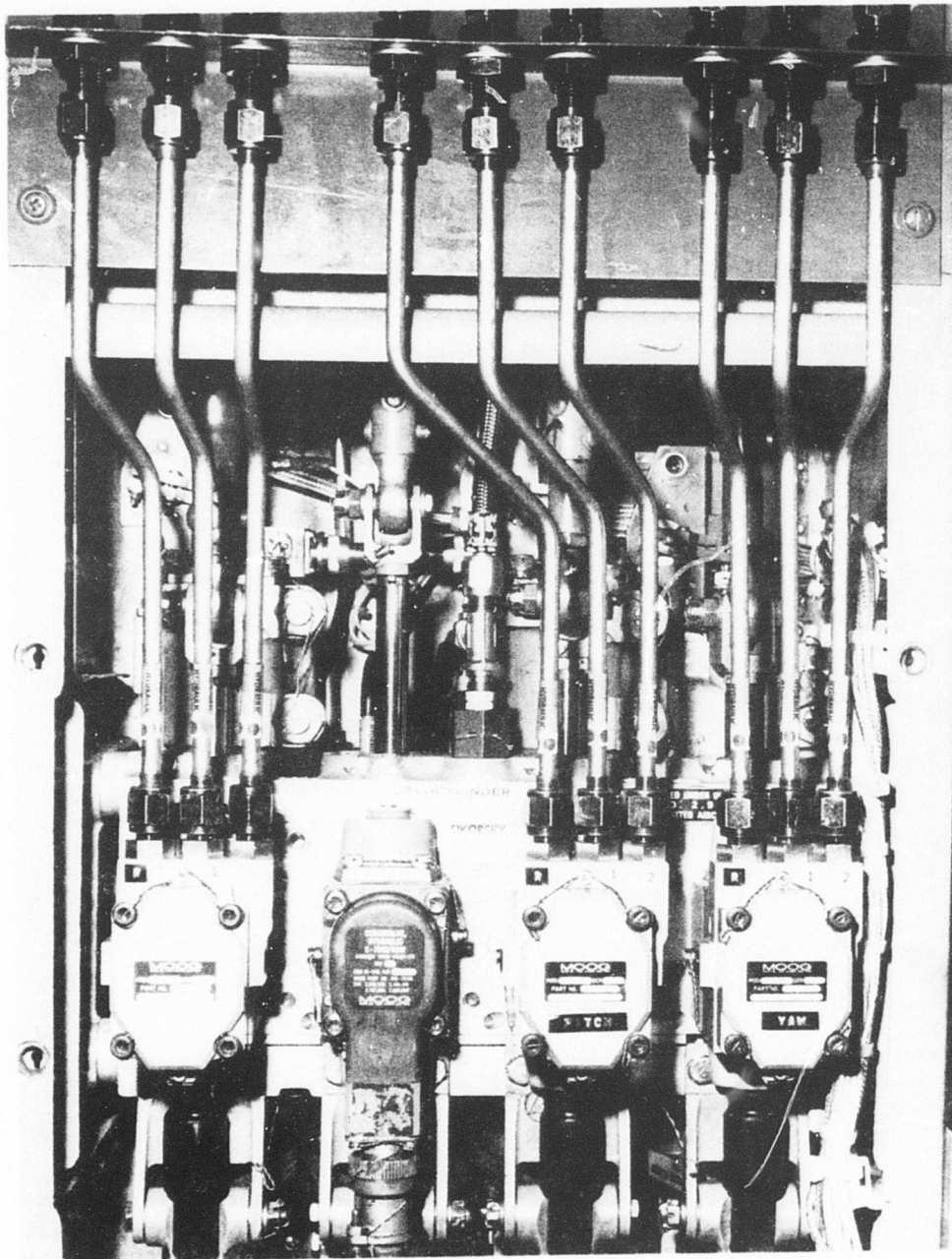


FIGURE 7. AFCS SERVO WITH HYDROFLUIDIC SERVO VALVES.

- . Cyclic and yaw trim control.
- . Introduction of series SAS signals into the controls.
- . Introduction of parallel attitude and heading hold signals into the controls.

The two-stage dual input servo valve that is utilized for both the SAS and pilot inputs was modified to accept hydrofluidic commands. Figure 7 is a picture of the AFCS servo as it was installed in the CH-54B helicopter. The first, third and fourth servo valves are the modified valves and the second is the conventional servo valve.

### Servo Operations

To see how this servo operates, we must first consider normal operation in response to a pilot applied input. Using Figure 8 as a guide, the pilot control stick input is at point A. For given pilot input motion, the rod at point A will move to the right, which will cause member B-C-D to rotate clockwise, pivoting at point B. The movement of member B-C-D will cause member D-E to move to the right, which will cause member E-F-G to rotate clockwise, pivoting at point F. The movement E-F-G will cause member G-H, which is the spool valve input rod, to activate the spool valve and move it to the left. This will crack the pressure port, thereby applying pressure to the left side of the power piston and causing it to move to the right. Assume the pilot now has stopped his input motion; member B-C-D will now rotate counterclockwise due to the power piston motion, causing point D to move to the left. This reversal of member movements will cause the spool valve to reverse and recenter. This action is referred to as mechanical feedback.

As previously noted, the spool valve unit of the servo controls the power piston; therefore, we can see that the SAS signals must also control this spool valve. Using Figure 8 as a guide, the operation of the servo valve is as follows: normally when there is no SAS error signal, the flapper will remain centered as shown in the drawing. When pressure is applied to the servo it will be felt at ports J and K. There will be an equal flow of fluid through each orifice, which will mean the pressure drop behind the orifices will be equal. This will develop an equal pressure behind each end of the spool valve. This equal pressure will keep the spool valve centered, as long as there are no other inputs. Now let us assume there is an input to the SAS error signal. This error signal will cause an unbalance in the servo valve, increasing the pressure at  $C_2$  and decreasing the pressure at  $C_1$ . This unbalance will cause the flapper to rotate clockwise, pivoting at point I. As the flapper rotates clockwise it tends to close orifice K, increasing the pressure behind this orifice, thereby increasing the pressure on the right side of the spool valve. As the flapper moves away from orifice J, it tends to open the orifice to a greater flow of fluid, which will decrease the pressure behind this orifice and decrease the pressure on the left side of the spool valve. This increase of pressure on the right side of the spool valve and decrease in pres-

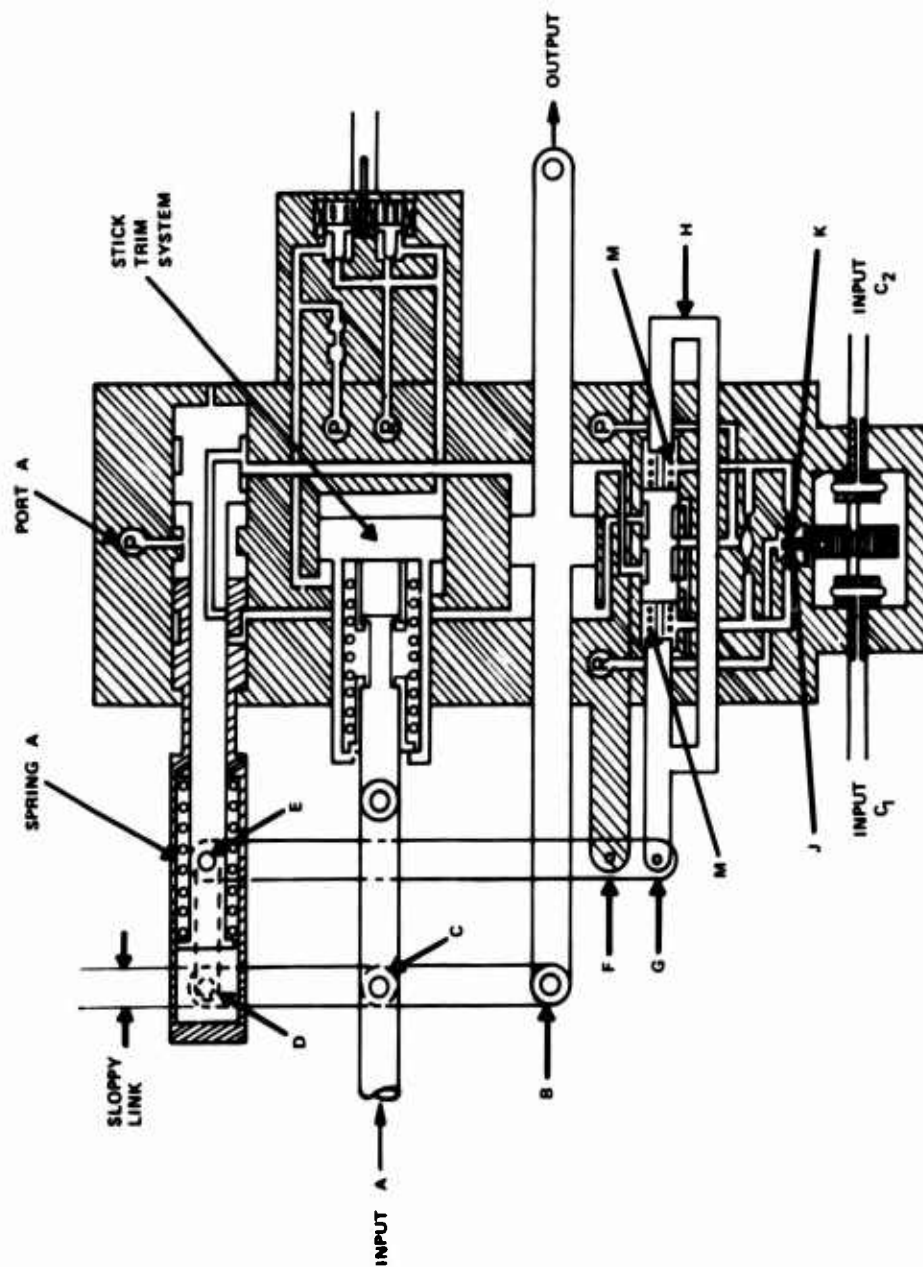


FIGURE 8. CH-54B AFCS SERVO.

sure on the left side of the spool valve will cause the spool valve to move to the left. The distance the spool valve moves to the left will depend on the difference in pressure between the two ends and the pressure on the spring behind the spool valve. When the difference in pressure is equalized by spring tension and the spool valve stops moving, the mechanical feedback, mentioned previously, will close the pressure port of the spool valve, stopping the power piston movement. As the error diminishes, the differential pressure at  $C_1$  and  $C_2$  will equalize and the compressed left spring will now force the spool valve to move to the right, returning to its original position. As the spool valve moves back to its original position the action just described reverses and the power piston also returns to its original position. The space "M" between the spool valve and the spool valve input rod will determine how far the ASE signal inputs can move the spool valve. This is how the ASE authority is limited.

The bypass actuator and slop eliminator are the parts of the servo unit that open and close the bypass port and lock and unlock the sloppy link which allows the pilot to fly through the servo when it is selected off.

When there is no pressure applied to the servo, the bypass ports are open and allow hydraulic fluid to flow freely from one side of the power piston to the other, in either direction. This action prevents a hydraulic lock of the power piston. At this time the sloppy link is locked and the power piston can be moved by a direct linkage to the flight control rods from the pilot's control sticks.

When pressure is applied to the servo the bypass ports close and hydraulically lock the servo. In this condition the spool valve must be moved in order to move the power piston. Under this condition the sloppy link is open which allows free movement of the spool valve by the flight control rods attached to the pilot's controls.

The operation is as follows: When pressure is applied to pressure port A, it will force the actuating piston to the right, opening the sloppy link, as shown in Figure 8. At the present time, this action will close the bypass port. When pressure is removed, spring A will expand, closing the sloppy link and opening the bypass port.

The trim features of the cyclic and yaw sections of the servo handle control position trim and serve as the means of introducing the attitude and heading hold signals.

## Electrical Design

### AFCS/SAS Amplifiers

The pitch, roll and yaw electronic SAS was disabled for the flight test by disconnecting the wiring to the AFCS servo. The oscillatory shut-off unit (OSU) was also deactivated. The OSU is an integral part of the basic SAS to detect any large amplitude midfrequency (2-25 Hz)

oscillatory SAS signals. If this condition should occur, the offending channel is automatically disconnected. Since an oscillatory failure is not possible in the fluidic SAS, this feature is not required.

### Control Panels

The HYSAS control panel is shown in Figure 9. The panel allows the pilot to select power for the HYSAS pump and also engage and disengage the HYSAS. The AFCS control panel is shown in Figure 10. This panel is the production panel used on the CH-54B aircraft. The engage logic was modified to interlock the engagement of AFCS #1 (attitude and heading) with the prior engagement of HYSAS. The "auto fail reset" feature which is controlled by the OSU was deactivated for the tests. AFCS #1 stick trim and yaw must be engaged to obtain attitude and heading hold.

Figure 11 is a sketch of the pilot and copilot cyclic stick grips, which supplied the main system quick-release functions. The cargo button (to the right of the trim switch) was used as the main release for the entire system. The "AFCS Servo Off" button, located on the left-forward portion of the cyclic, could also release the system by disengaging the entire AFCS servo. However, this feature is only momentary and a permanent disengagement must be accomplished on the main AFCS control panel. The trim release on the left center side of the cyclic will release the attitude hold in a momentary manner.

Figure 12 is an electrical schematic of the AFCS engage sequence. AC power is supplied to the pump thru one set of contacts of relay K<sub>1</sub>. The HYSAS is engaged by pressing the engage button energizing relay K<sub>1</sub>. This interlocks the HYSAS engage through the three release buttons, engages the HYSAS hydraulic pump, and arms the AFCS engage line. If any of the normally closed switches are opened, K<sub>1</sub> will drop off the line, releasing the HYSAS and AFCS.

The AFCS servo is released by holding either normally closed switch or by releasing the servo engage switch.

### Safety Considerations

The fluidic SAS was designed to have the same limited authorities as the approved system. Therefore, if a hardover condition had occurred the pilot could overcome the malfunction in the same manner as the certified aircraft. The electronic attitude and heading hold parallel servo system had a limited velocity, which is easily recovered by the pilot with a maximum of 8 pounds in cyclic and 80 pounds in yaw. Table 2 depicts the authorities on the production CH-54B and the aircraft modified for HYSAS. The rates and authorities are the same.

The system was certificated by the FAA to FAR 29 paragraph 29.1239 and Advisory Circular 29-1 at the envelope extremes including flight to 16,000 feet.

TABLE 2. AFCS AUTHORITIES		
	CH-54B	CH-54B (HYSAS)
<b>HYSAS</b>		
PITCH	$\pm 10\%$	$\pm 10\%$ (SAME STOPS)
ROLL	$\pm 10\%$	$\pm 10\%$ (SAME STOPS)
YAW	$\pm 5\%$	$\pm 5\%$ (SAME STOPS)
<b>ATTITUDE-HEADING</b>		
PITCH	10%/SEC	10%/SEC (UNCHANGED)
ROLL	10%/SEC	10%/SEC (UNCHANGED)
YAW	10%/SEC	10%/SEC (UNCHANGED)

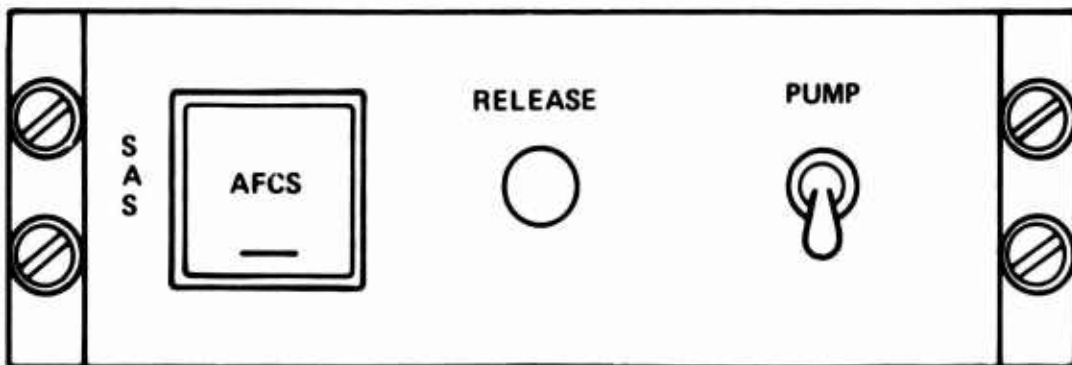


FIGURE 9. HYSAS CONTROL PANEL.

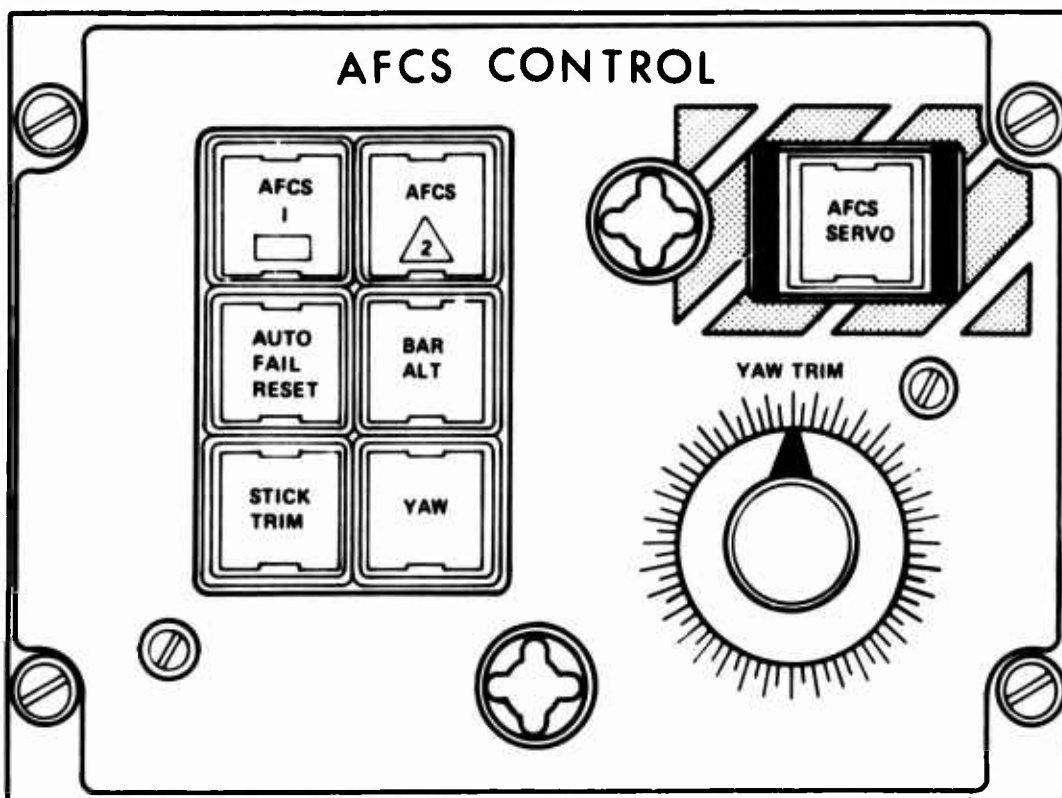
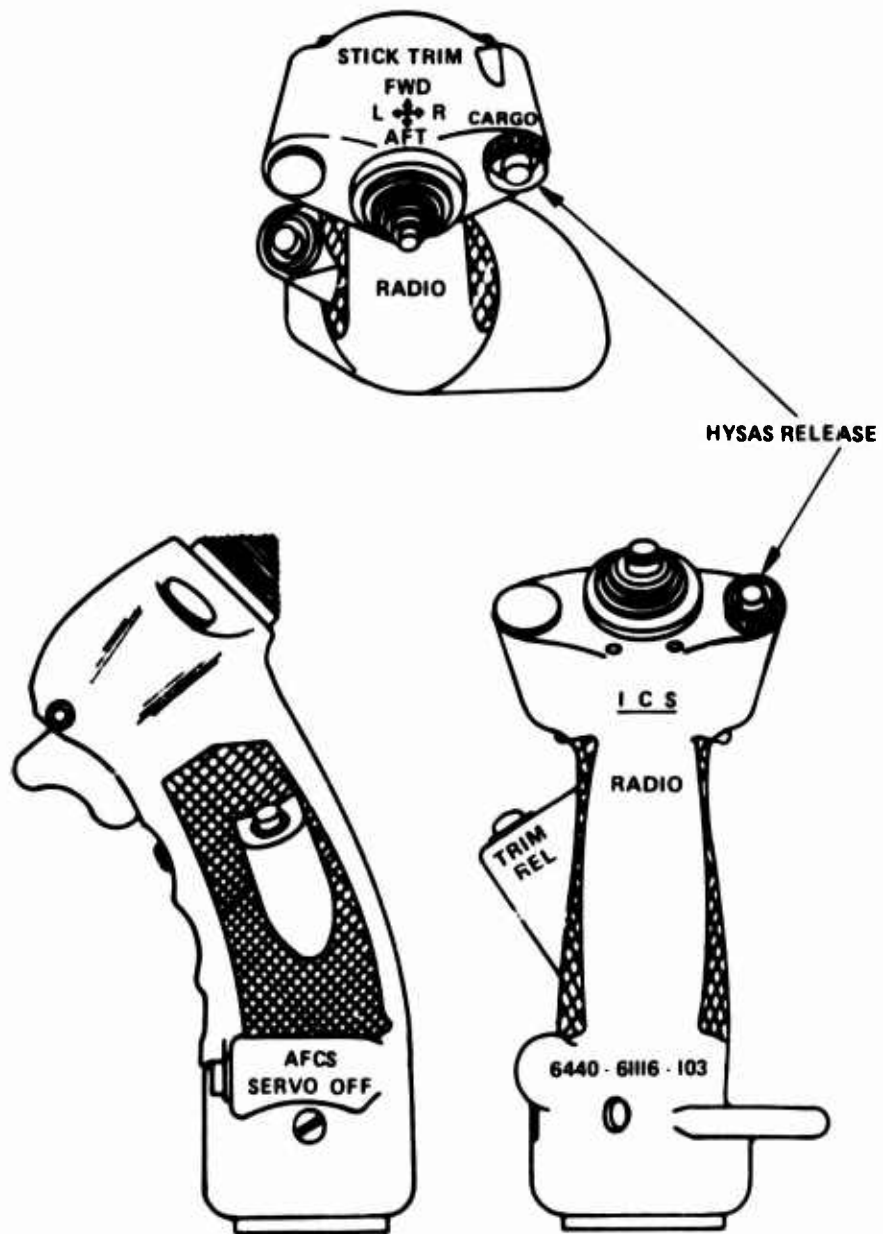


FIGURE 10. AFCS CONTROL PANEL.



**FIGURE II. FORWARD CYCLIC STICK GRIPS.**

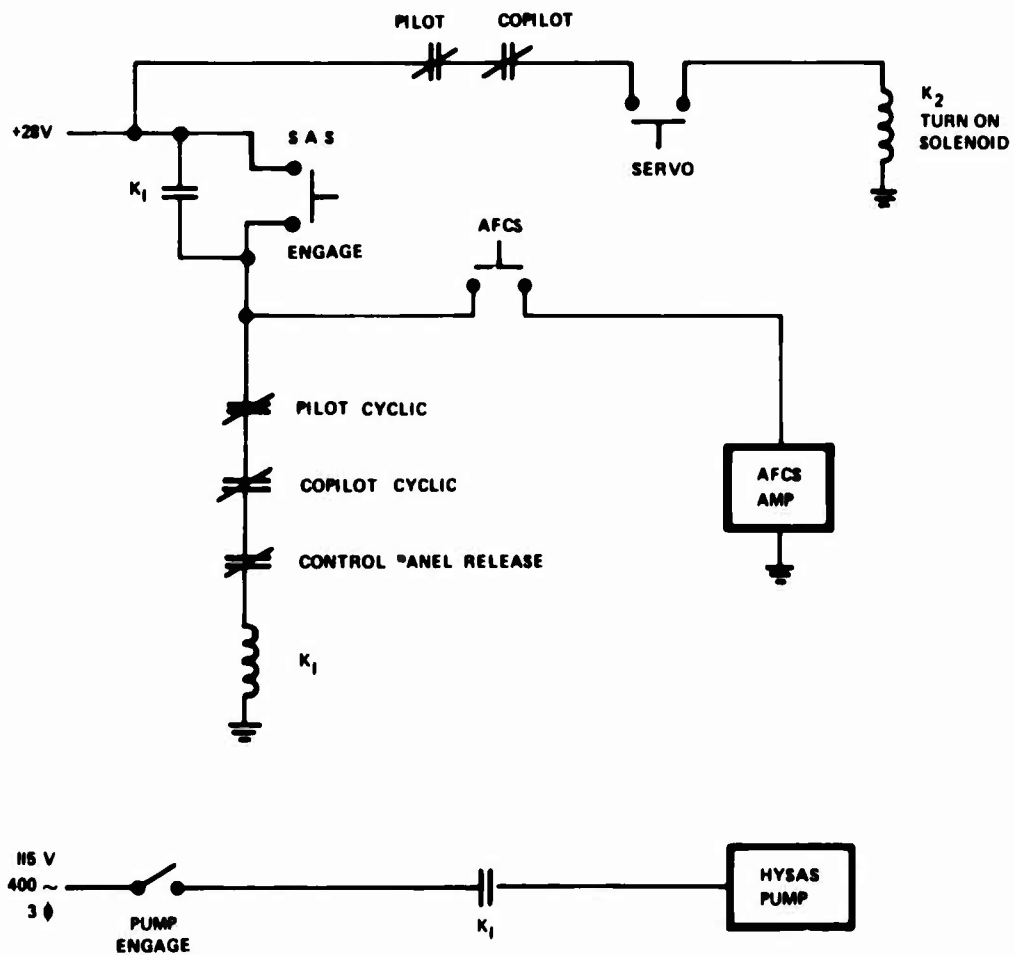


FIGURE 12. AFCS ENGAGE LOGIC.

## SYSTEM INTEGRATION TESTS

System integration tests were performed in the laboratory to confirm satisfactory operation prior to installation and flight test. Servo valve testing, system frequency response, threshold, noise, temperature and flow tests were used to evaluate the fluidic components.

### Servo Valve Evaluation

Figures 13, 14 and 15 depict the pitch, roll and yaw HYSAS servo valve linearity and threshold.

The zero based linearity is within 2.5% in the worst case, which is quite comparable to the production electromechanical valve. The threshold of the roll valve, which demonstrated the largest threshold, was found to be 0.025 PSI.

Figure 16 depicts the typical frequency response of the hydrofluidic servo valve. The response is good to 4 Hz with a slight resonance noted at 2 Hz. It was concluded that the design and operation of the servo valve were adequate for flight test.

### Frequency Response

Figures 17, 18 and 19 depict the pitch, roll and yaw HYSAS frequency response with the output measured at the power piston of the modified CH-54B AFCS servo. While the amplitude ratios for the three axes are close to the theoretical values, the phase response rolls off quicker than predicted. The roll-off of the phase curve for all three SAS's was attributed to the emptying time of the vortex rate sensor (transport lag). The roll-off appeared to be identical to that obtained under Contracts DAAJ02-68-C-0039 and DAAJ02-69-C-0036. This is caused by the transport lag being 45 msec rather than 20 msec, as assumed for the theoretical calculations.

Since the identical roll axis was flown by Sikorsky in an S-58T in December 1971 it was assumed that the effect of the roll-off at the higher frequencies would be minimal.

### System Noise and Threshold

Table 3 itemizes the results of the threshold and noise laboratory tests. The threshold falls between a quarter and a third of a degree per second. This was considered adequate for flight test and comparable to a conventional electromechanical rate gyroscope.

The roll output noise was considered excellent. The pitch axis was considered satisfactory and the yaw axis marginal at 1.2% peak to peak. Concern was raised more at the frequency than the amplitude, as the low frequency was very close to the main rotor RPM. However, due to the state of the art of that particular set of fluidics, it was decided that no major improvements could be accomplished.

TABLE 3. NOISE AND THRESHOLD MEASUREMENTS			
	PITCH	ROLL	YAW
NOISE, output Peak to Peak	0.65% @ 3.9 Hz	NEG	1.2% @ 3.9 Hz
THRESHOLD	0.234°/sec	0.294°/sec	0.25°/sec

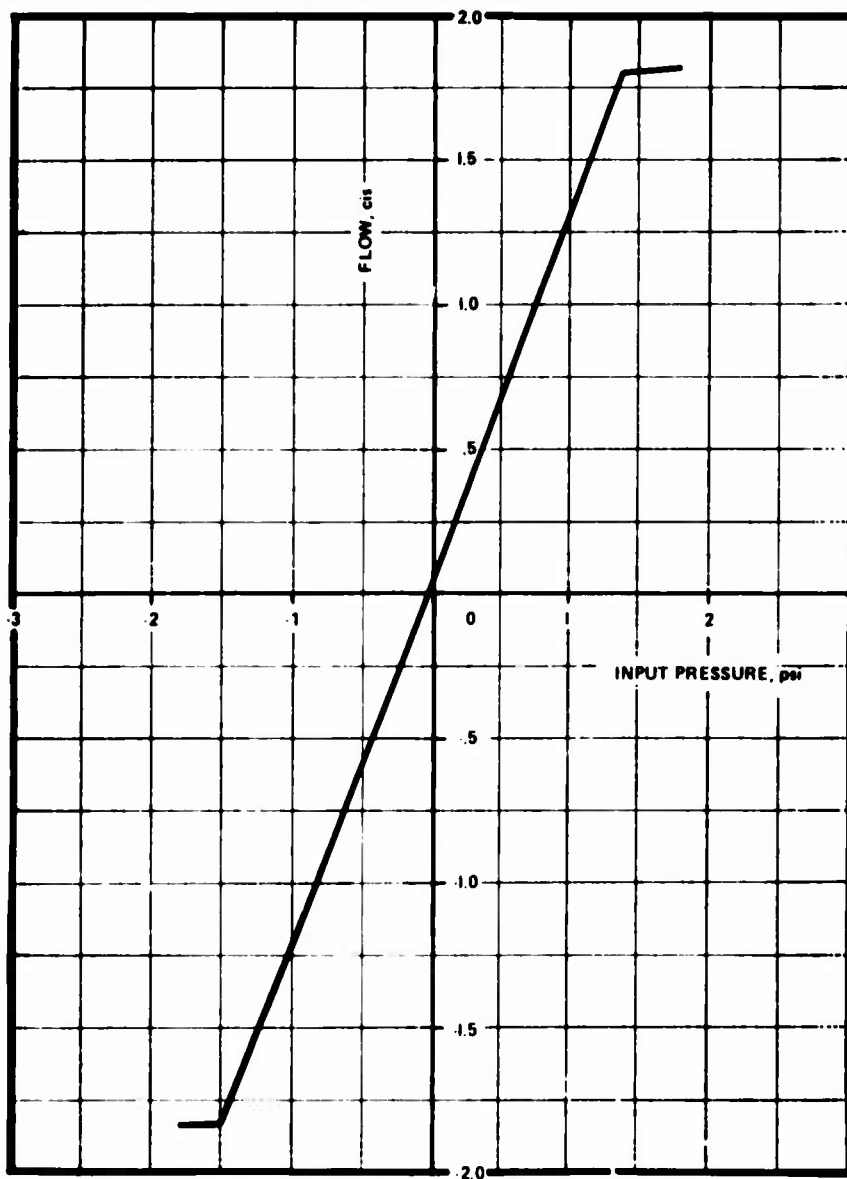


FIGURE 13. PITCH SERVO VALVE LINEARITY.

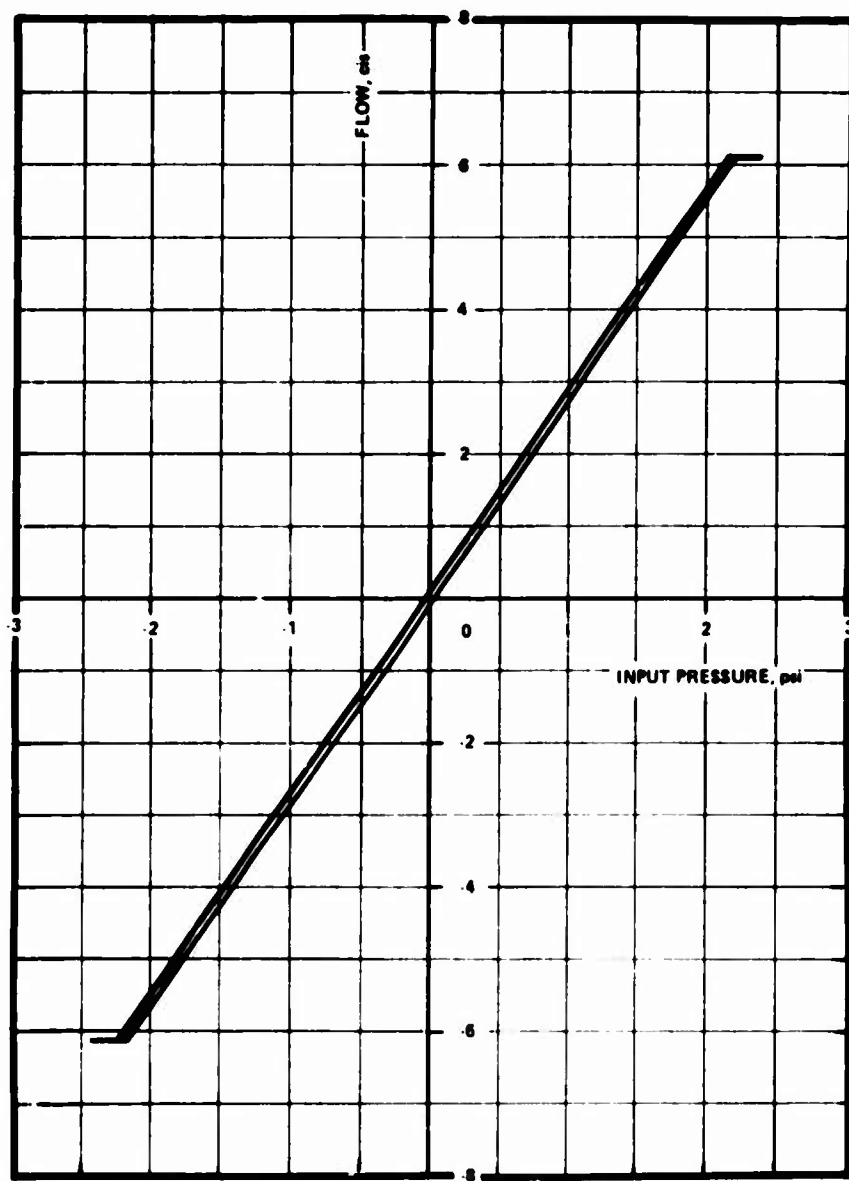


FIGURE 14. ROLL SERVO VALVE LINEARITY.

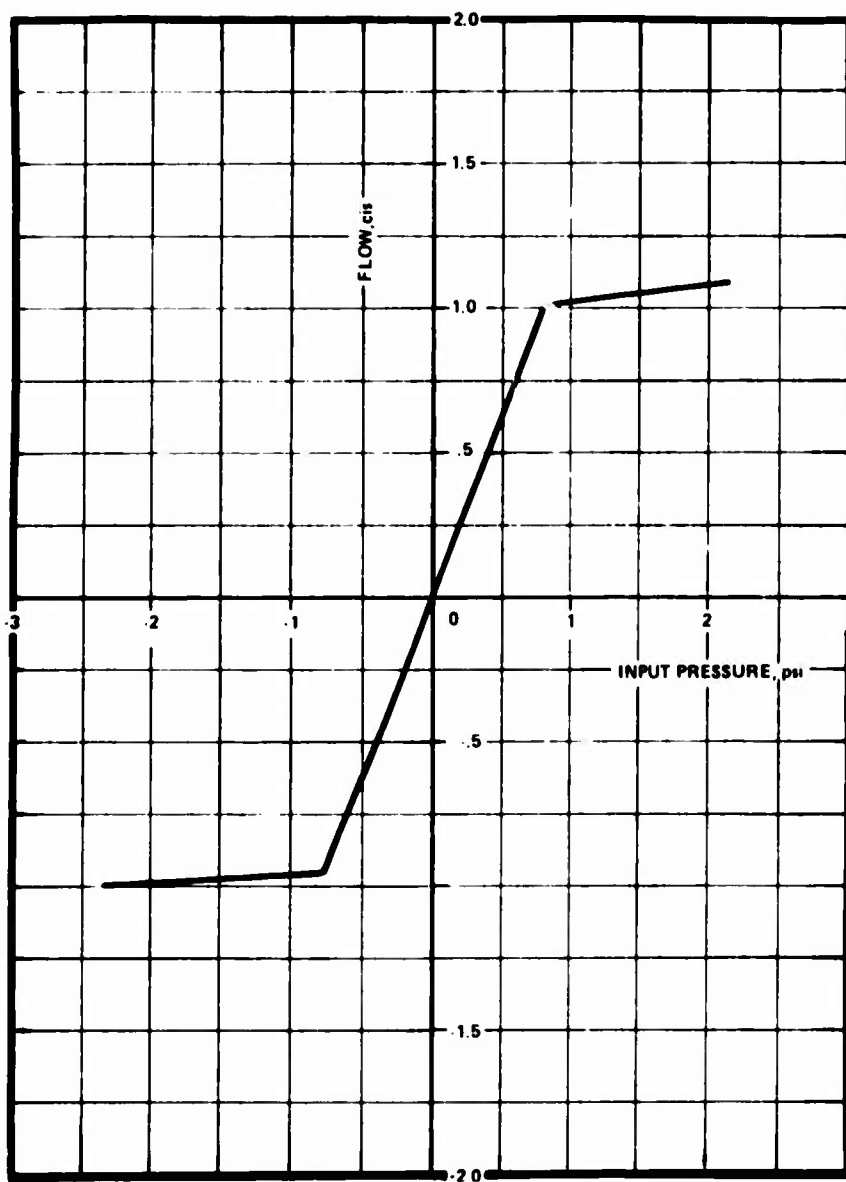


FIGURE 15. YAW SERVO VALVE LINEARITY.

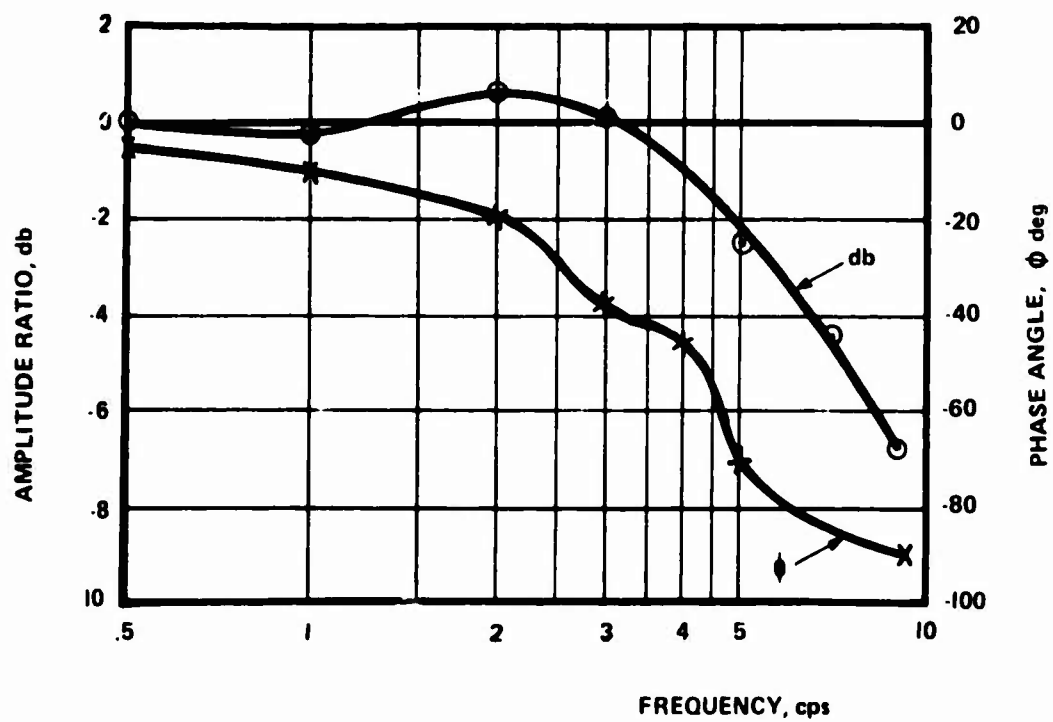


FIGURE 16. HYDROFLUIDIC SERVO VALVE FREQUENCY RESPONSE .

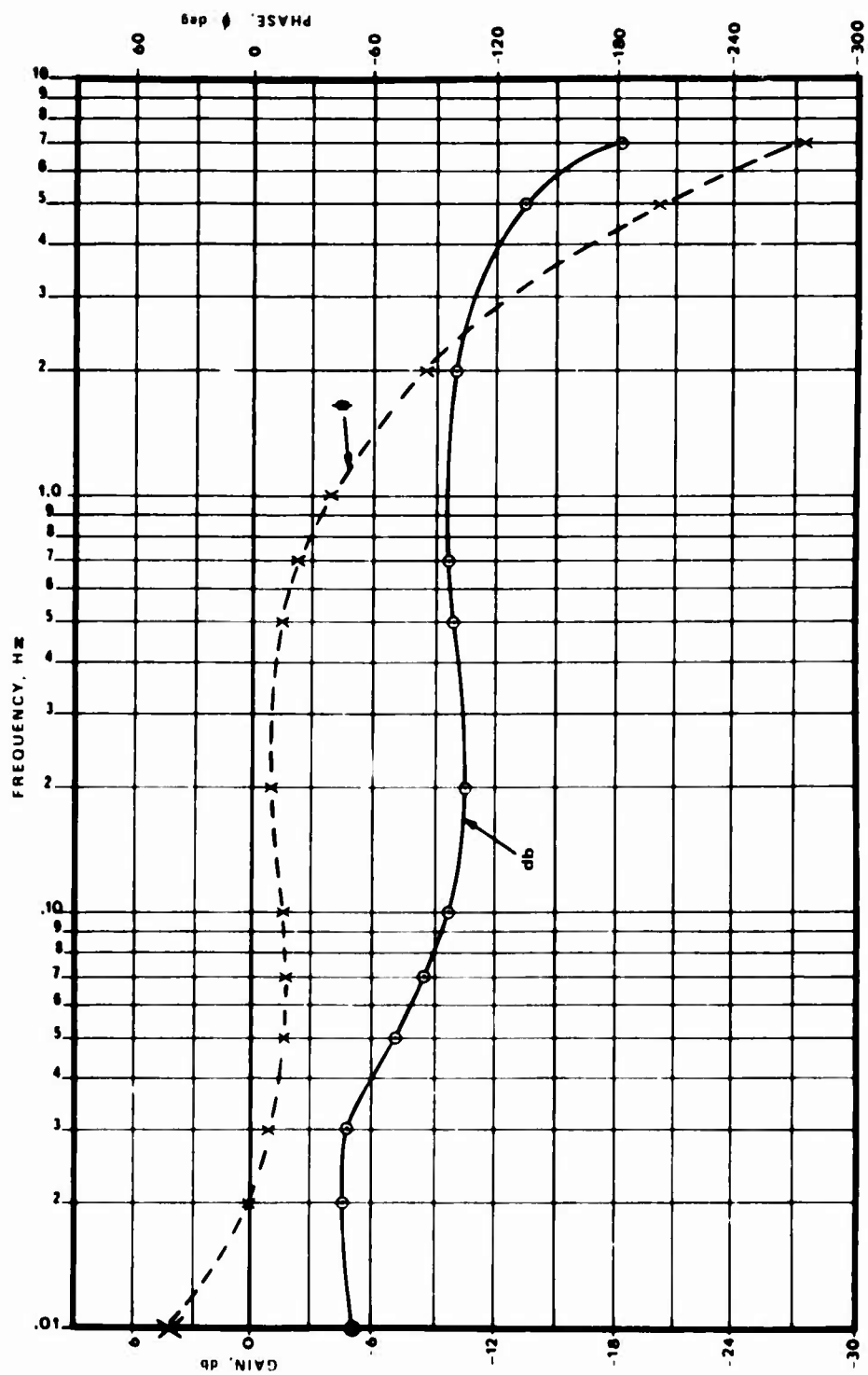


FIGURE 17. CH-54B PITCH AXIS SAS DYNAMIC RESPONSE TEST RESULTS (OUTPUT MEASURED AT SERVOACTUATOR).

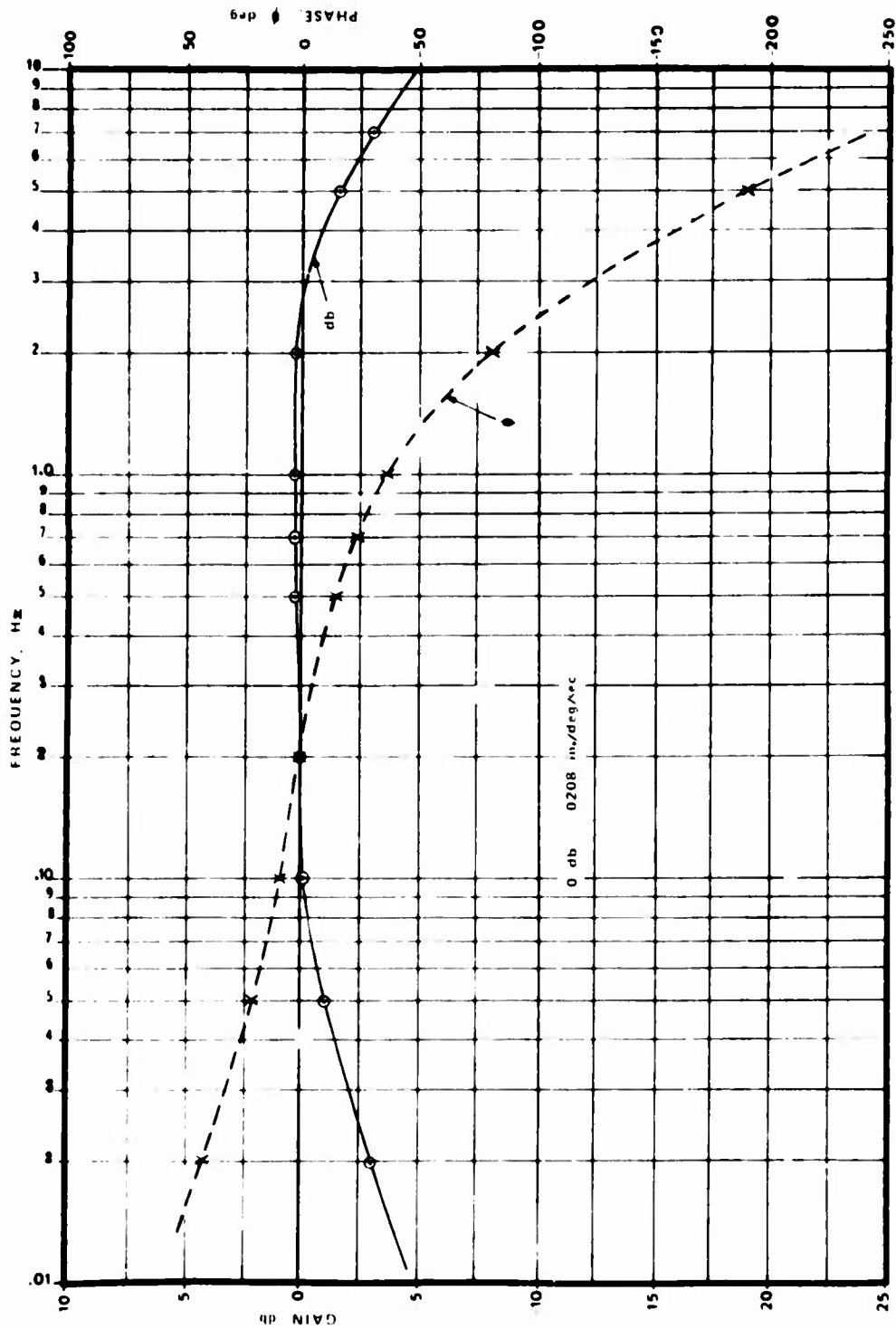


FIGURE 18. CH-54B ROLL AXIS SAS DYNAMIC RESPONSE TEST RESULTS (OUTPUT MEASURED AT SERVOACTUATOR).

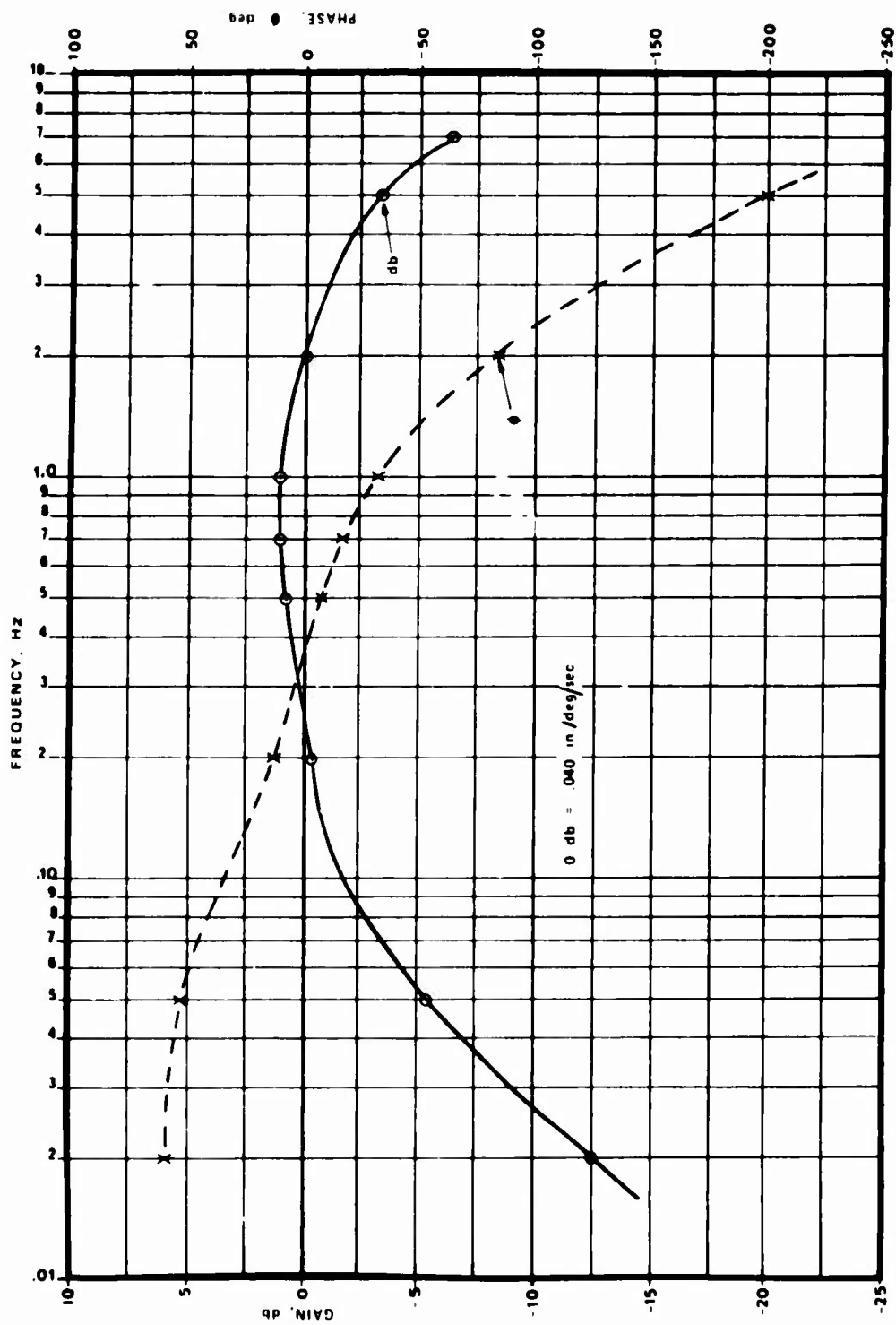


FIGURE 19. CH-54B YAW AXIS SAS DYNAMIC RESPONSE TEST RESULTS (OUTPUT MEASURED AT SERVOACTUATOR).

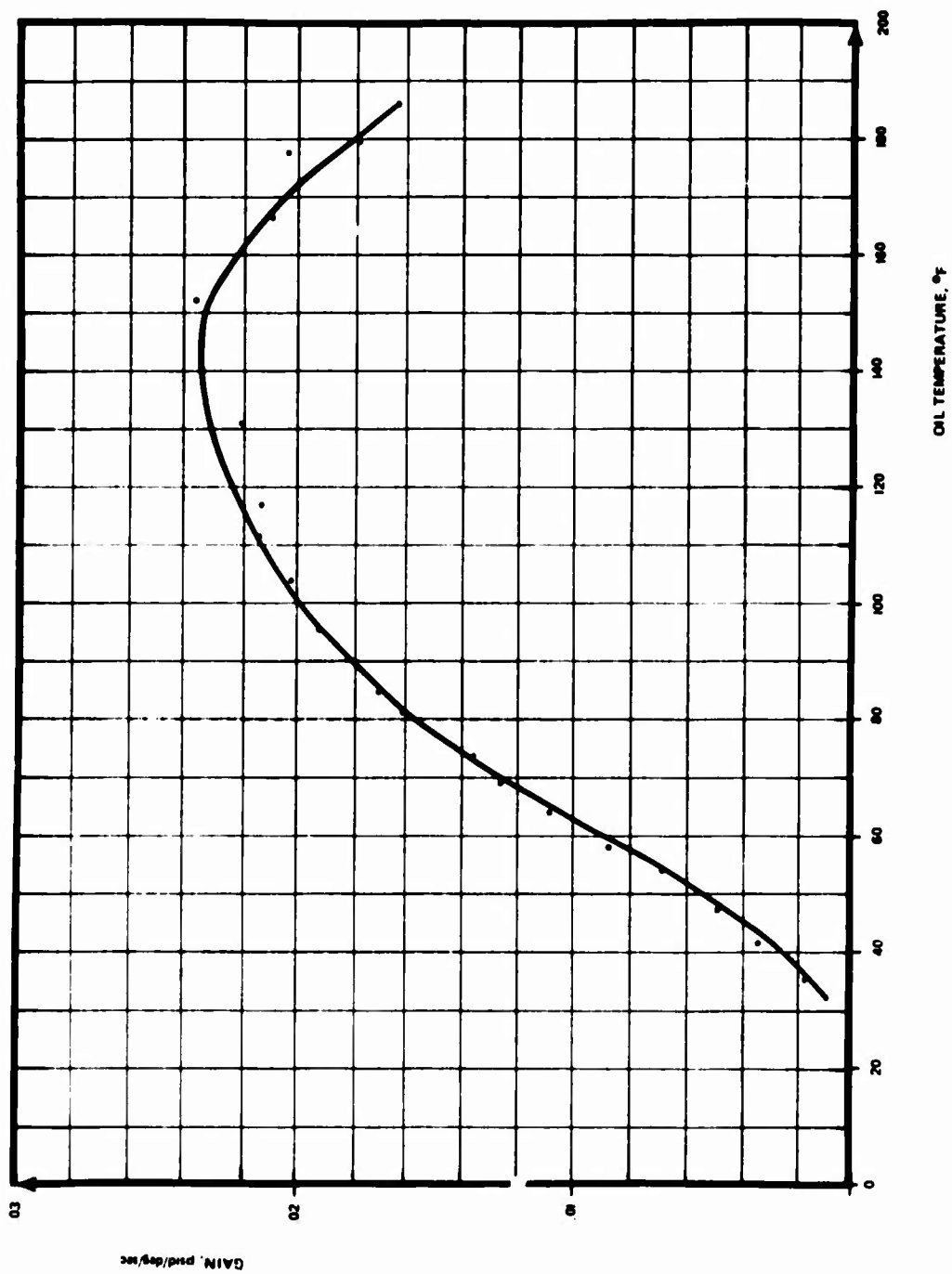


FIGURE 20. GAIN VERSUS OIL TEMPERATURE.

### System Temperature Effects

The gain variation with temperature tests were run at Sikorsky using the identical roll channel. Gain measurements were made at various oil temperatures with the following parameters held constant:

Back Pressure	-	165 PSI
Indicated Flow	-	2.3 GPM
Table Amplitude	-	$\pm 10^\circ$
Table Frequency	-	0.2 Hz

The curve shown in Figure 20 of gain (reflected at the servo input) vs temperature probably represents a worse case, because the flowmeter readings were not corrected for temperature. Although no temperature data is readily available, it is likely that with the higher viscosity of cold oil the flowmeter would read high. This would cause the low gain to be the combined effect of low temperature and low flow. The temperature characteristics of the flow regulator used in an actual application become especially critical. Since the stabilized hydraulic oil temperature on the CH-54B is above  $140^\circ\text{F}$ , an integral hydraulic system with a heat exchanger was used for flight tests.

### Flow Effects

The transport lag time decreased with increased flow rate, since the lag is basically the time required for the fluid to move through the rate sensor. The transport lag varied as shown in Figure 21. The transport lag could be a potential problem. For good stability the transport lag should be minimized. The 45 msec lag in the unit tested results in a  $45^\circ$  phase shift at 3 Hz. This lag can be reduced at the cost of increased flow, as noted in Figure 22, but with a corresponding penalty in power.

Figure 22 depicts the variation in gain with a change in flow. Since the relationship is reasonable for a 1-3 gpm change, it was decided to use this phenomenon for fine tuning the gain during flight test.

### Problems Encountered During Test

During the testing, the pitch servo valve failed. The servo was apparently subjected to a high transient load between the control input and the control reference pressure lines which bent the armature in the servo valve. The reason for the transient load is unknown, but is surmised to have been large air bubbles in the system at pump start-up. This condition would not occur in a hard system in a helicopter.

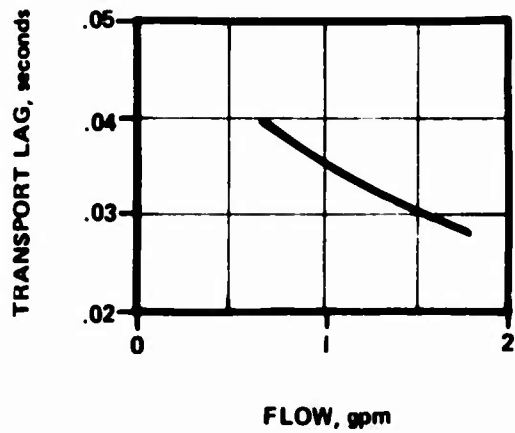
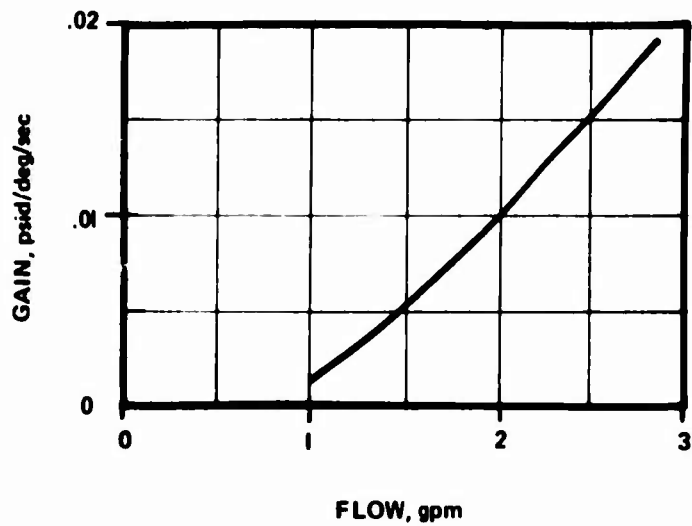


FIGURE 21. TRANSPORT LAG  
VERSUS FLOW.

FIGURE 22. GAIN VERSUS FLOW.



## INSTALLATION

Figure 23 is a sketch showing the overall layout of the system installed in the left rear of the CH-54B cabin. The rear pilot's seat and collective control were removed to provide the required space for the HYSAS equipment and the Brush recorder. The HYSAS has its own hydraulic power source, so the basic aircraft's hydraulic system remains unchanged.

### Aircraft Interface

Figure 24 is a diagram showing the HYSAS interface with the aircraft mechanical, hydraulic and electric system. The shaded portion depicts the HYSAS addition to the aircraft. The main philosophy of the installation was to replace the electronic SAS with HYSAS without modifying any of the mechanical or hydraulic control system; therefore, the failure modes are unchanged. Figure 24 shows that the continuity of the control system remains unchanged. The HYSAS components used were supplied GFE and had been previously proven in extensive lab tests under Contracts DAAJ02-68-C-0039 and DAAJ02-69-C-0036 as well as 60 hours of flight test on a UH-1 under Contract DAAJ02-70-C-0017.

Figure 25 is a picture of the HYSAS component installation in the test aircraft. The hydraulic pump and heat exchanger can be seen in the foreground. The main flow control valve is to the right of the pump and was used to modify the gain of all three HYSAS channels. The pitch channel is located in the center and the roll channel is behind the heat exchanger. The yaw channel is directly behind the roll channel. The gain adjust manifold is located behind the pitch axis and was used to change the channel gains  $\pm 30\%$ .

Figure 26 is a picture of the modified CH-54B AFCS servo installed in the test aircraft. Bleed ports for each channel are located above the servo as noted. These were utilized to remove the air entrapped in the system. The modified servo valves and their respective plumbing can be observed in the foreground. The relatively long run of lines did not have any measurable effect on system performance.

Figure 27 is a picture of the center console of the test aircraft. The locations of the HYSAS and AFCS control panels are shown; since the cargo release switch was used as the main system release, the cargo release control had to be activated prior to flight. This control is shown in the lower right of the figure.

### Problems Encountered During Ground Test

Excessive system noise caused by air entrained in the hydraulic fluid was the major problem encountered during the system check-out. Air entrainment was aggravated by the fact that the reservoir in this system is very small and does not act as a collecting point for bubbles. Flow charging the system from a large auxiliary reservoir finally debubbled the entire system.

During the check-out, the system gains were found to be higher than those obtained in the laboratory as a result of shorter hydraulic power line

lengths used in the helicopter. This problem was solved by installing a valve in the power line to throttle the flow to the lab flow rates (2.08 gpm).

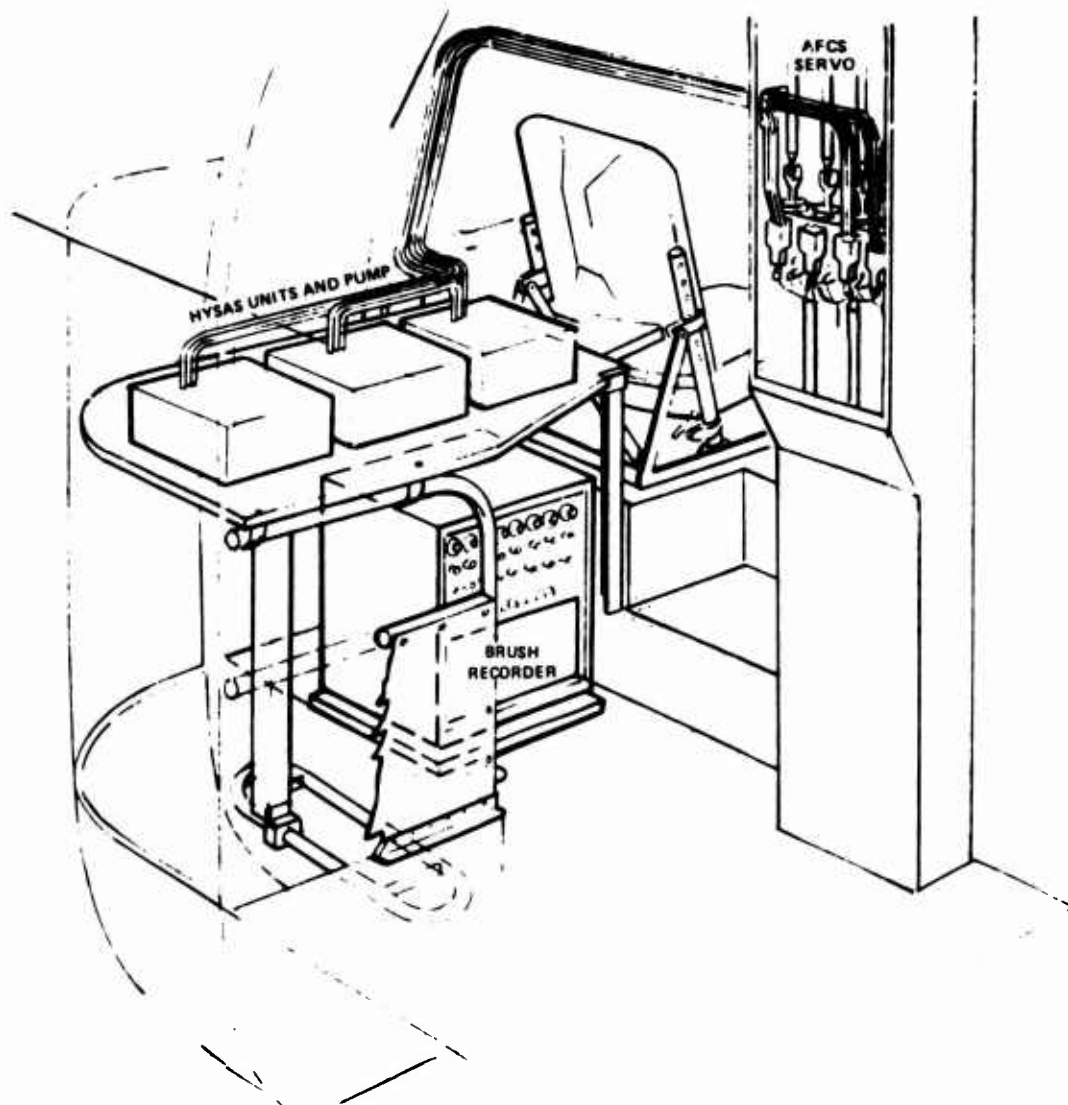


FIGURE 23. HYSAS INSTALLATION.

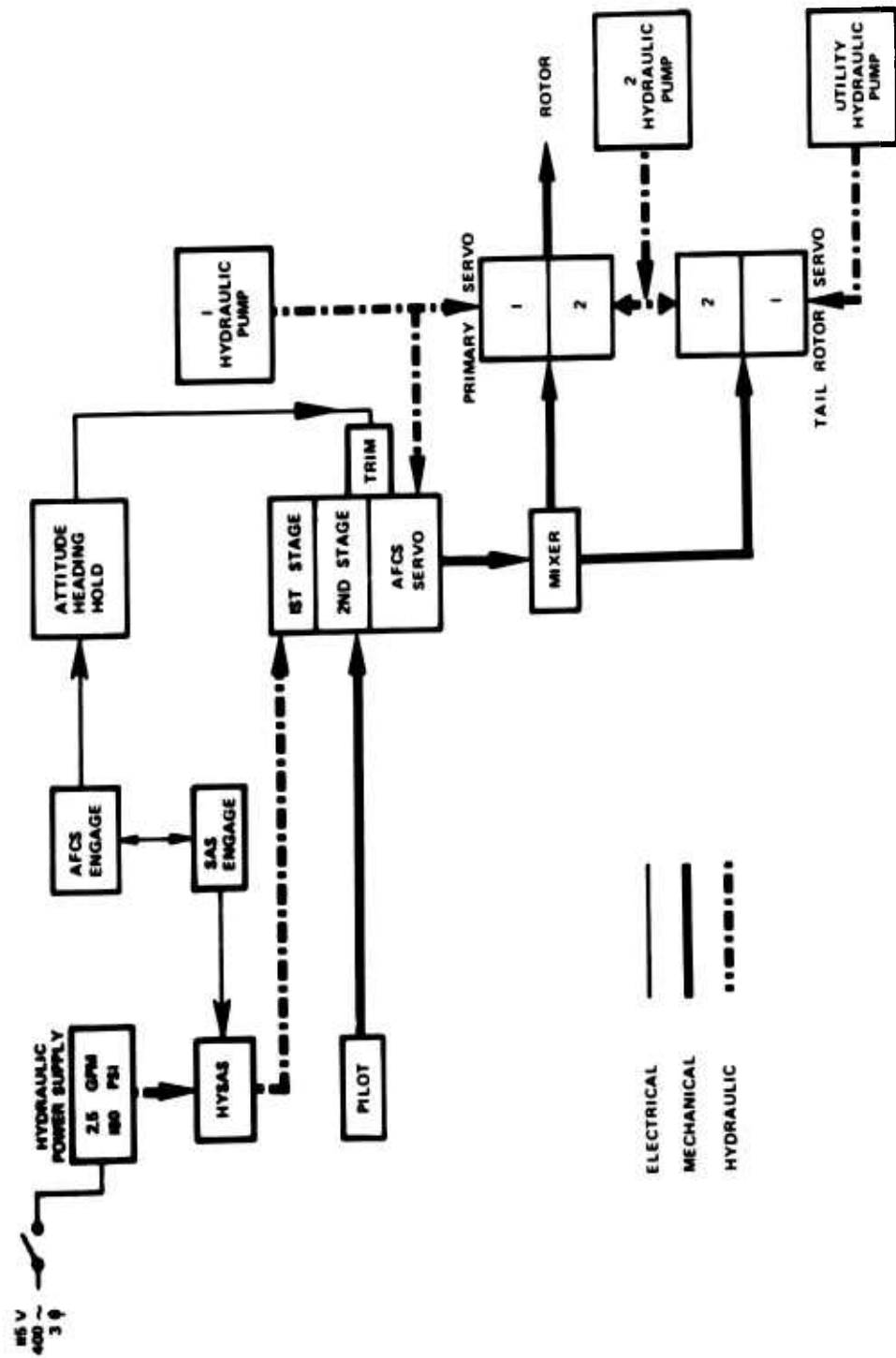


FIGURE 24. HYSAS SCHEMATIC.

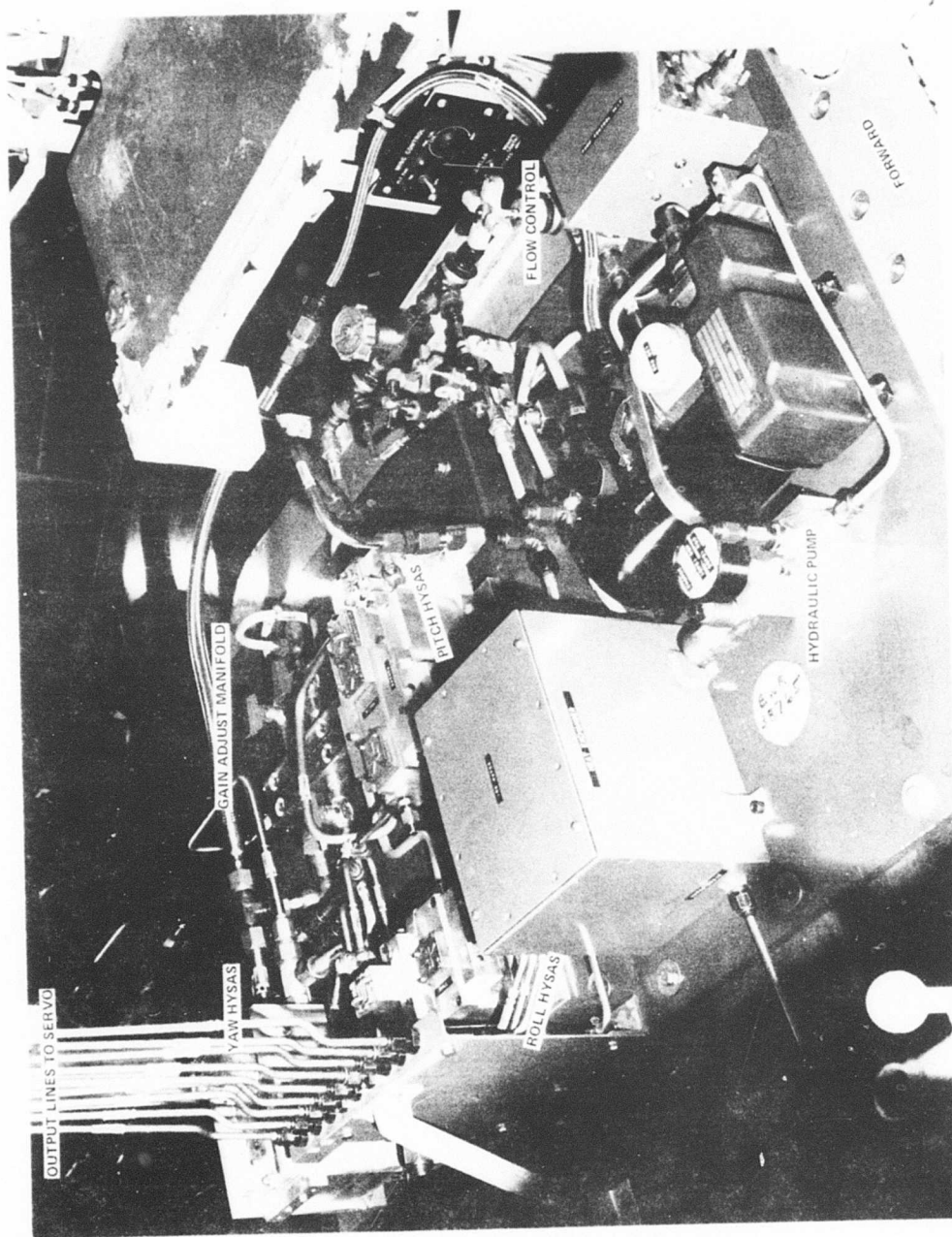


FIGURE 25. HYSAS COMPONENT INSTALLATION.

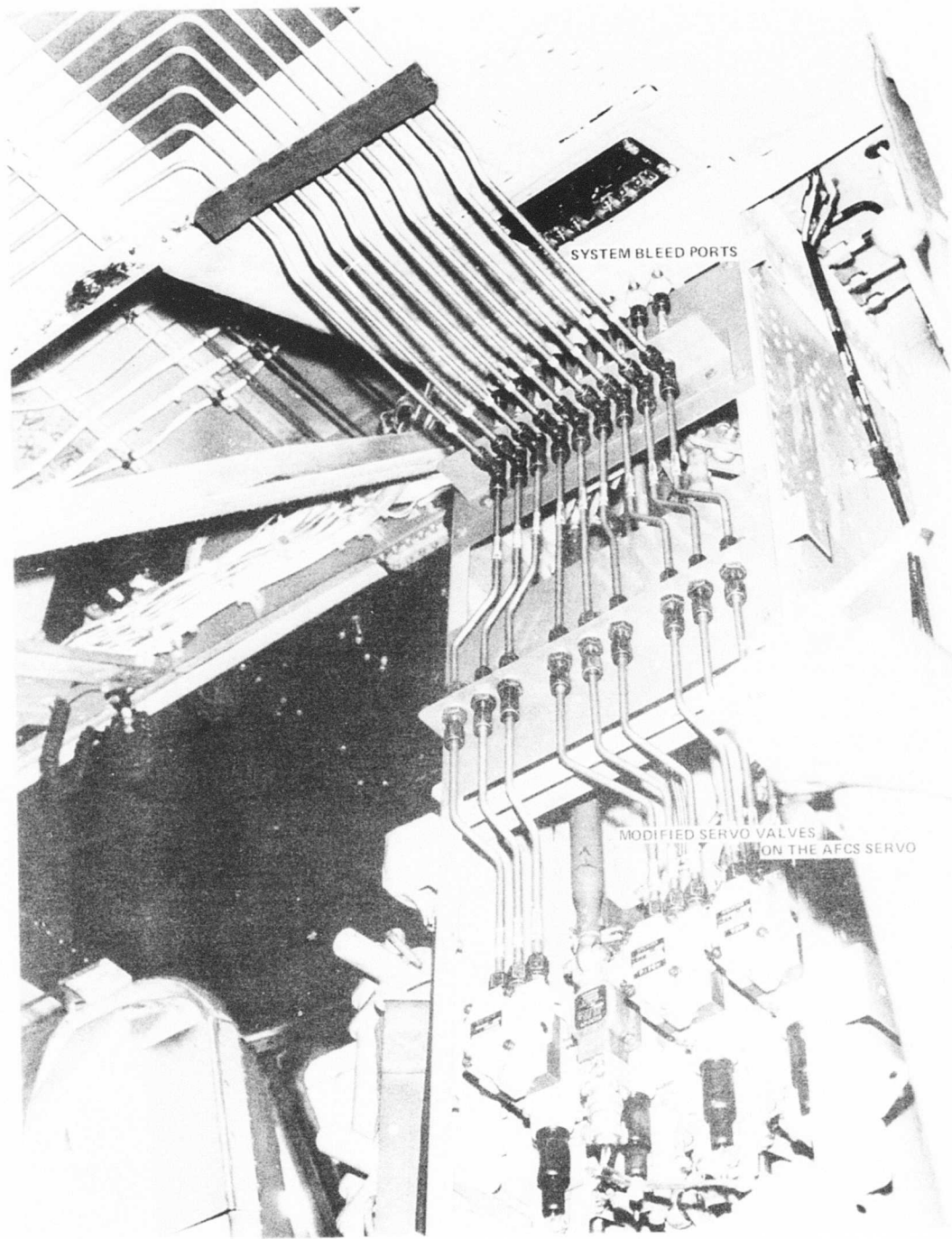


FIGURE 26. AFCS SERVO INSTALLATION.

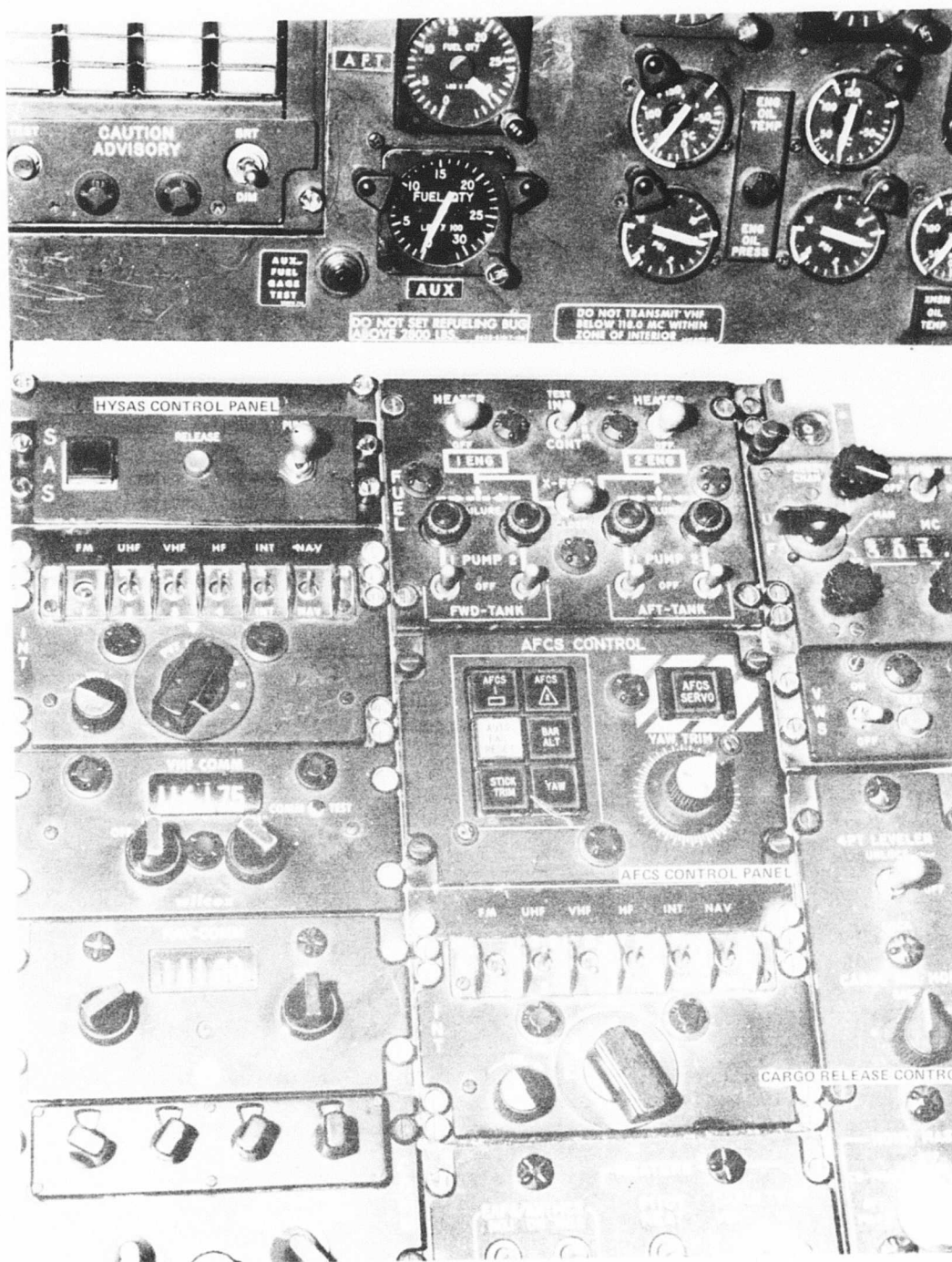


FIGURE 27. CH-54B CENTER CONSOLE.

## TEST AND EVALUATION

Seven flights and one ground run-taxi were used to evaluate the system. The system was totally operational during all of the aircraft tests and never sustained a malfunction. The gains of the HYSAS required some adjustment as well as the gains of the heading hold system. The interaction of the gain adjustments caused difficulty in arriving at precise settings. Four flights were used to set the gains and three flights were used to evaluate the operation and record data. Table 4 is a summary of the individual axis damping for a pulse input at all of the flight conditions evaluated. Figures 37 to 87 are the actual time histories recorded.

### PITCH AXIS

The pitch channel displayed reasonable attitude hold. The system damping is highlighted in Table 4. The pilots felt that more damping was required for maneuvering flight; however, the final pitch HYSAS gain was reduced from the simulation gain by approximately 15% because the pitch HYSAS responded to the aircraft's first vertical bending mode, which created an instability. This phenomenon is caused by the rate sensor being mounted at a point on the airframe that is moving at an angle (node) as the fuselage bends. The HYSAS detects the angular rate and computes a signal to correct for the motion, resulting in the instability. This problem may have been corrected by mounting the pitch vortex rate sensor at an anti-node which is located 35 feet aft of the actual location. However, it was felt that this was not in the scope of the program. When an abrupt entry into autorotation was accomplished, the pitch HYSAS saturated, allowing the nose to pitch down. This phenomenon is normal to the CH-54B with the production SAS. The final gains used for the flight test are noted below:

$$\text{PITCH HYSAS} \quad \frac{B_{1s}}{\dot{\theta}_1} = 0.774 \left( \frac{2.12 s + 1}{8 s + 1} \right) \left( \frac{10s}{10 s + 1} \right) e^{-0.45s} \quad \%/\text{deg/sec} \quad (7)$$

$$\text{ATTITUDE HOLD} \quad \frac{B_{1s}}{\theta_1} = (0.02 s + .85) \%/\text{deg} \quad (8)$$

### ROLL AXIS

The roll channel displayed excellent attitude hold. The system damping is highlighted in Table IV. The final roll HYSAS gain was increased from the simulation gain by 23% as a result of pilot comments after maneuvering flight. This increase was actually somewhat higher than desired due to the coarse means of adjusting gains. This gain occasionally resulted in a phenomenon known as a "glitch". The "glitch" is usually encountered in forward flight when the aircraft is rapidly displaced from trim and allowed to return. The rate gain is high enough to stop the aircraft from rolling before the aircraft returns to trim and then the system must start motion in the same direction to achieve trim. The pilots felt that the damping was

TABLE 4. DAMPING ACHIEVED AFTER PULSE INPUTS				
AXIS	FLIGHT CONDITIONS - DAMPING			LOADING
	HOVER	80 KNOTS	V MAX	
PITCH	0.5	1.0	0.8	C G 335 G W 28,200
ROLL	1.0	1.0	1.0	
YAW	0.4	0.5	0.5	
PITCH	0.7	0.8	0.9	C G 336 G W 35,000
ROLL	1.0	1.0	1.0	
YAW	0.4	0.6	0.5	
PITCH	0.5	0.7	0.7	C G 346 G W 47,000
ROLL	0.4	0.8	0.8	
YAW	0.5	0.6	0.6	

slightly high, but the system compared favorably with the electronic SAS. The final gains used for the flight test are noted below:

$$\text{ROLL HYSAS} \quad \frac{A_{IS}}{\dot{\phi}} = 0.208 \left( \frac{10S}{10S + 1} \right) e^{-0.45S} \%/\text{deg/sec} \quad (9)$$

$$\text{ATTITUDE HOLD} \quad \frac{A_{IS}}{\dot{\phi}} = (0.13S + 1.25) \%/\text{deg} \quad (10)$$

#### YAW AXIS

The yaw channel displayed fair heading hold. The system damping is highlighted in Table 4 and is not quite as good as the analytical model. The main reason for this is that the gain of the HYSAS is approximately 10% lower than the gain required by the simulation. As noted in the test section, the yaw channel did exhibit a marginal amount of noise in the low frequency band. The gain that was flown was the maximum that could be tolerated due to the noise feedback to the airframe. Collective step inputs in a hover resulted in heading transients of no greater than 5 degrees, which is comparable to the production system. Autorotation entries in forward flight resulted in heading transients of 5 degrees, which was also comparable to the production system. The outer loop rate gain was increased to account for the yaw washout and the lower inner loop gain. The final gains used for flight test are noted below:

$$\text{YAW HYSAS} \quad \frac{\Theta_{TR}}{\dot{\psi}} = 0.473 \left( \frac{2S}{2S + 1} \right) e^{-0.45S} \%/\text{deg/sec} \quad (11)$$

$$\text{HEADING HOLD} \quad \frac{\Theta_{TR}}{\dot{\psi}} = \left( 0.888S + 5.0 + \frac{0.1}{S} \right) \%/\text{deg/sec} \quad (12)$$

#### MANEUVERING FLIGHT

The following maneuvers were used to evaluate the system at the specified flight conditions:

- Hovering and S turns
- Sideward flight
- Rolling reversals ( $\pm 40^\circ$  attitude) with coordinated climbs and dives

The overall consensus of the pilot opinion was that the system performed well, but could use more gain in the pitch and yaw channels. Figure 87 depicts how well the system held attitude and heading in slight to moderate turbulence.

### CONCLUSIONS

The flight test program results demonstrated that a hydrofluidic SAS can adequately augment the stability of a large single-rotor helicopter. The HYSAS provided adequate damping and stabilization to allow the rate limited outer loop electronic system to work in excellent harmony. Since most helicopters display weak lateral-directional characteristics, a coordinated turn mode is a prerequisite of SAS and should be developed for HYSAS. The following specific conclusions are presented:

- . No major modifications are required for a hybrid fluidic inner loop and an electronic outer loop automatic flight control system.
- . The aircraft, as modified, meets the same IFR as the production CH-54B.
- . Sensor/control units will require mounting in remote locations to obviate interference with the bending modes of the fuselage of large helicopters.
- . A fluidic coordinated turn is required to improve the forward flight lateral-directional handling qualities.

## APPENDIX I

### SIMULATION DATA

This appendix contains hybrid simulator recordings taken at various speeds, gross weights and centers of gravity. The simulation was a full six degree total force and moment simulation.

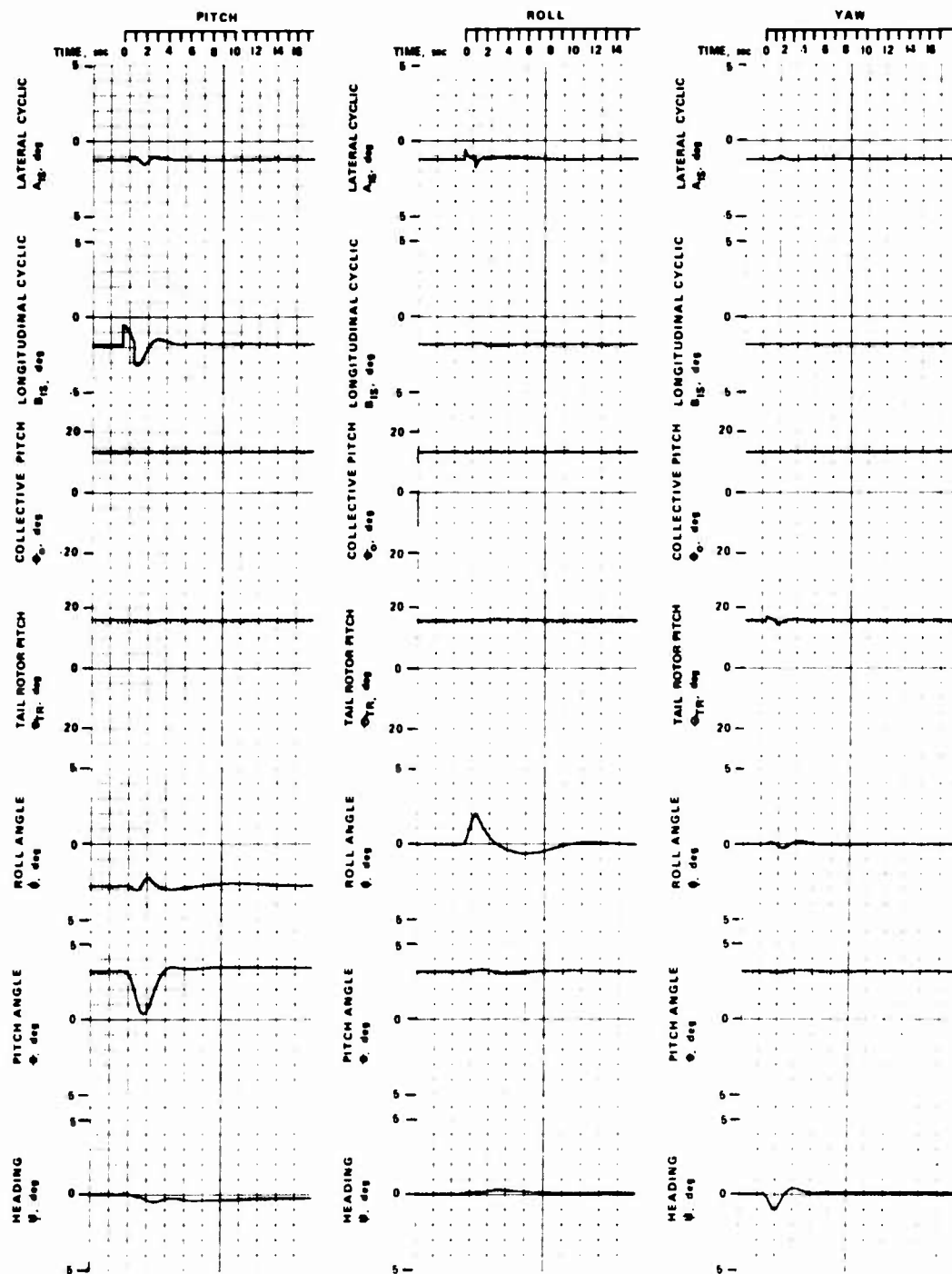


FIGURE 28. AIRCRAFT RESPONSE TO 5% - 1 SECOND CONTROL INPUTS  
FOR GW = 42,000 LB, CG = 336 IN. AT HOVER.

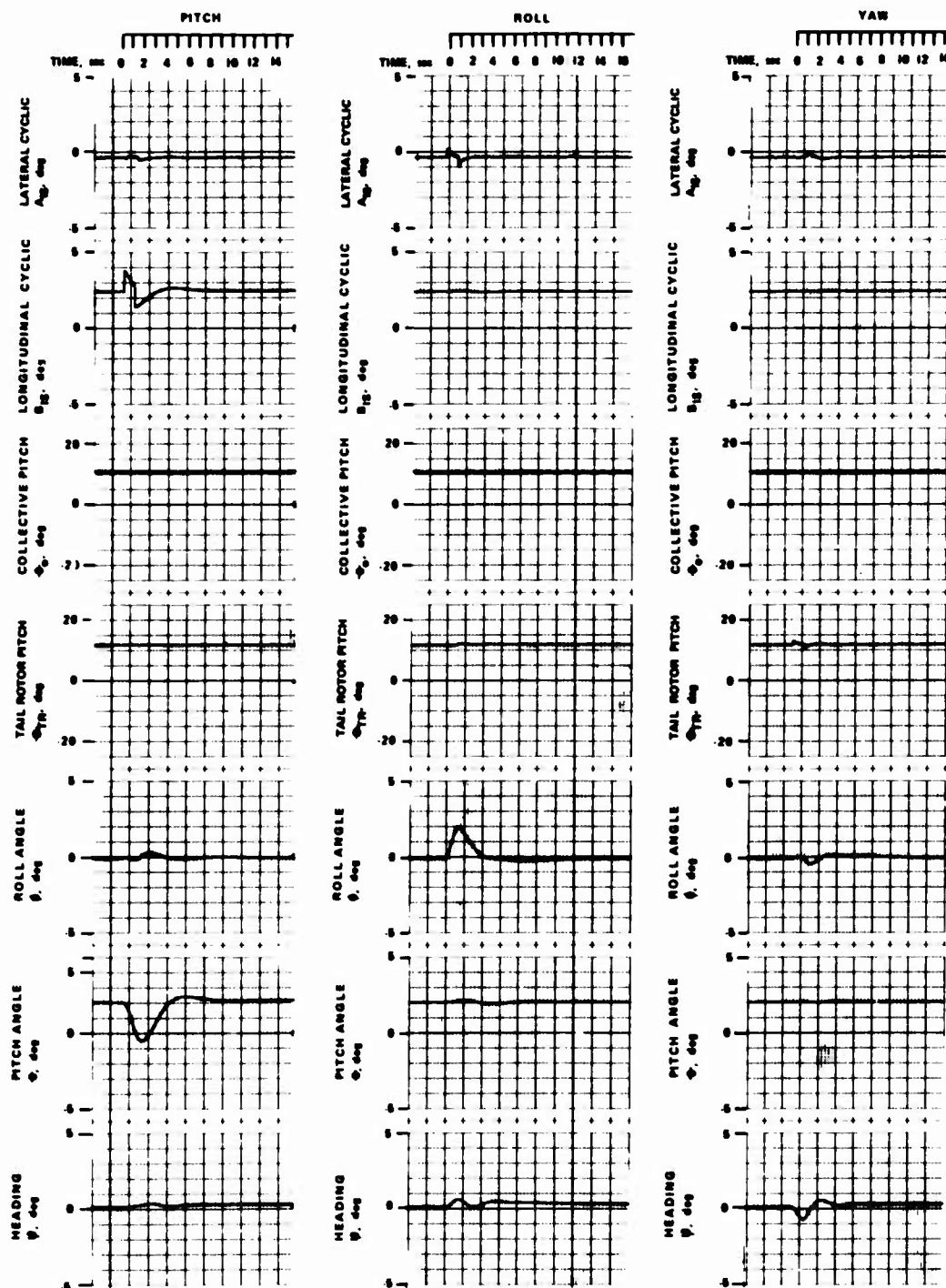


FIGURE 29. AIRCRAFT RESPONSE TO 5% - 1 SECOND CONTROL INPUTS FOR GW = 42,000 LB, CG = 336 IN. AT 80 KNOTS.

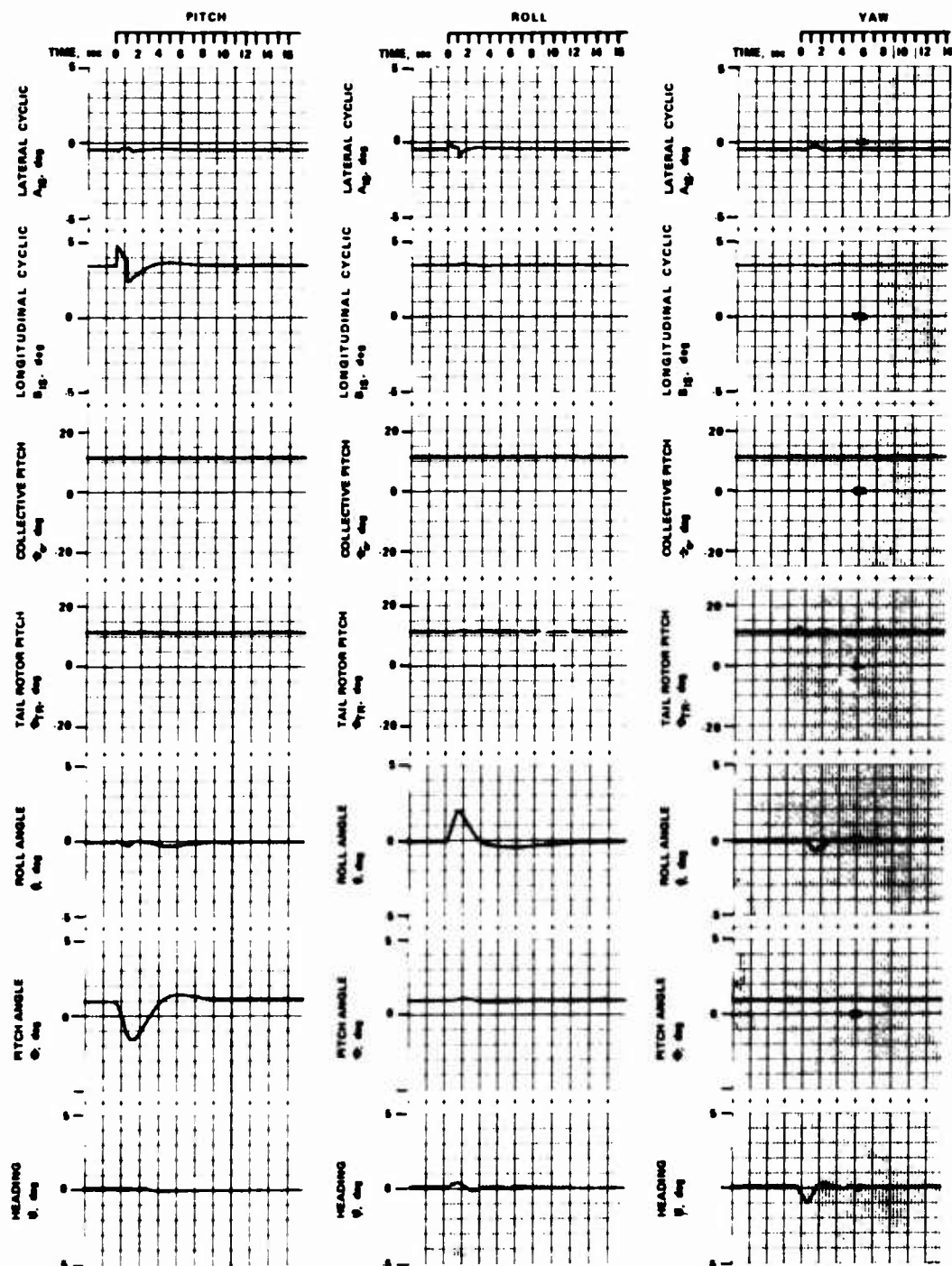


FIGURE 30. AIRCRAFT RESPONSE TO 5% - 1 SECOND CONTROL INPUTS  
FOR GW = 42,000 LB, CG = 336 IN. AT 100 KNOTS.

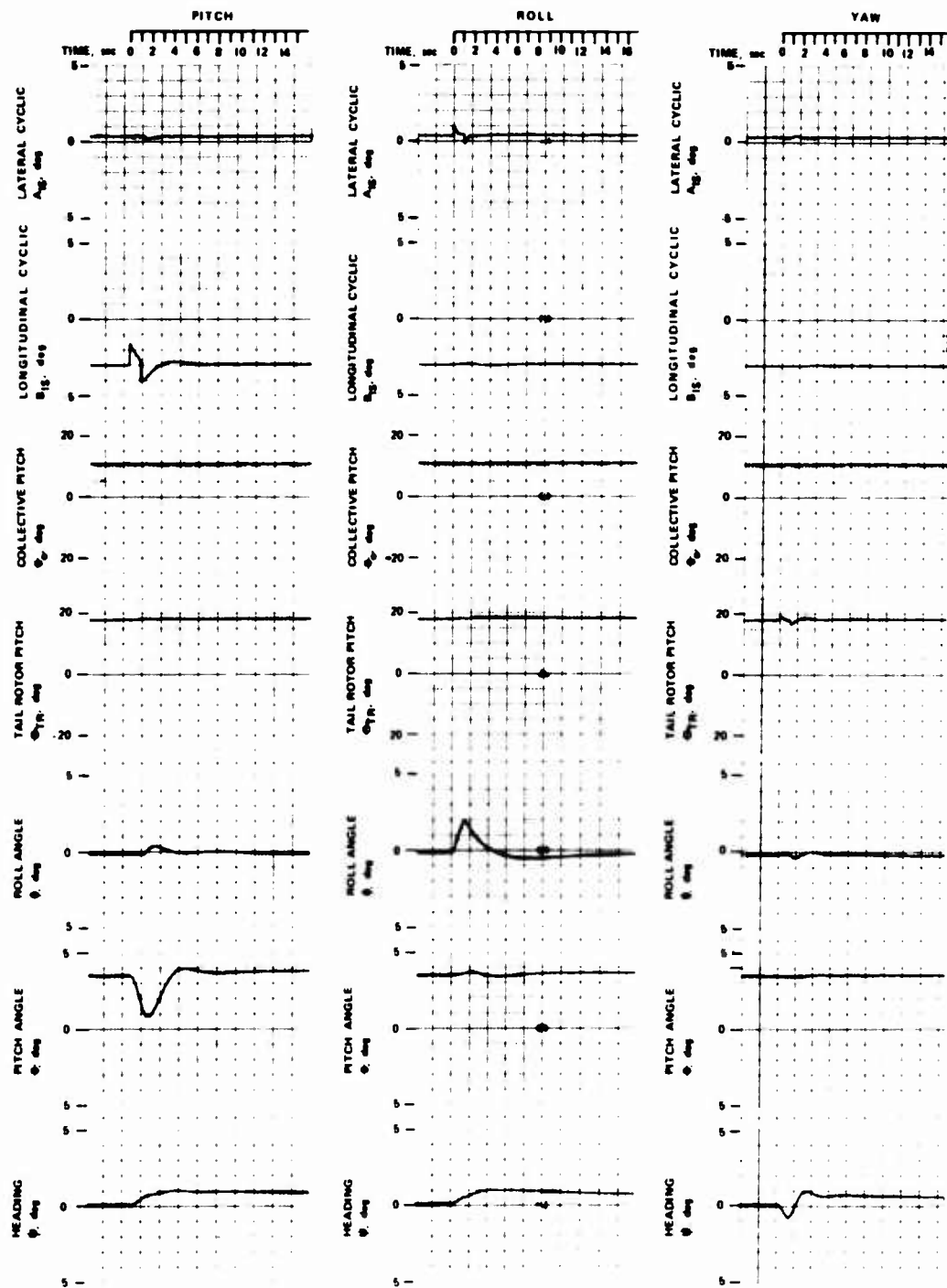


FIGURE 31. AIRCRAFT RESPONSE TO 5% - 1 SECOND CONTROL INPUTS  
FOR GW = 34,000 LB, CG = 328 IN. AT HOVER.

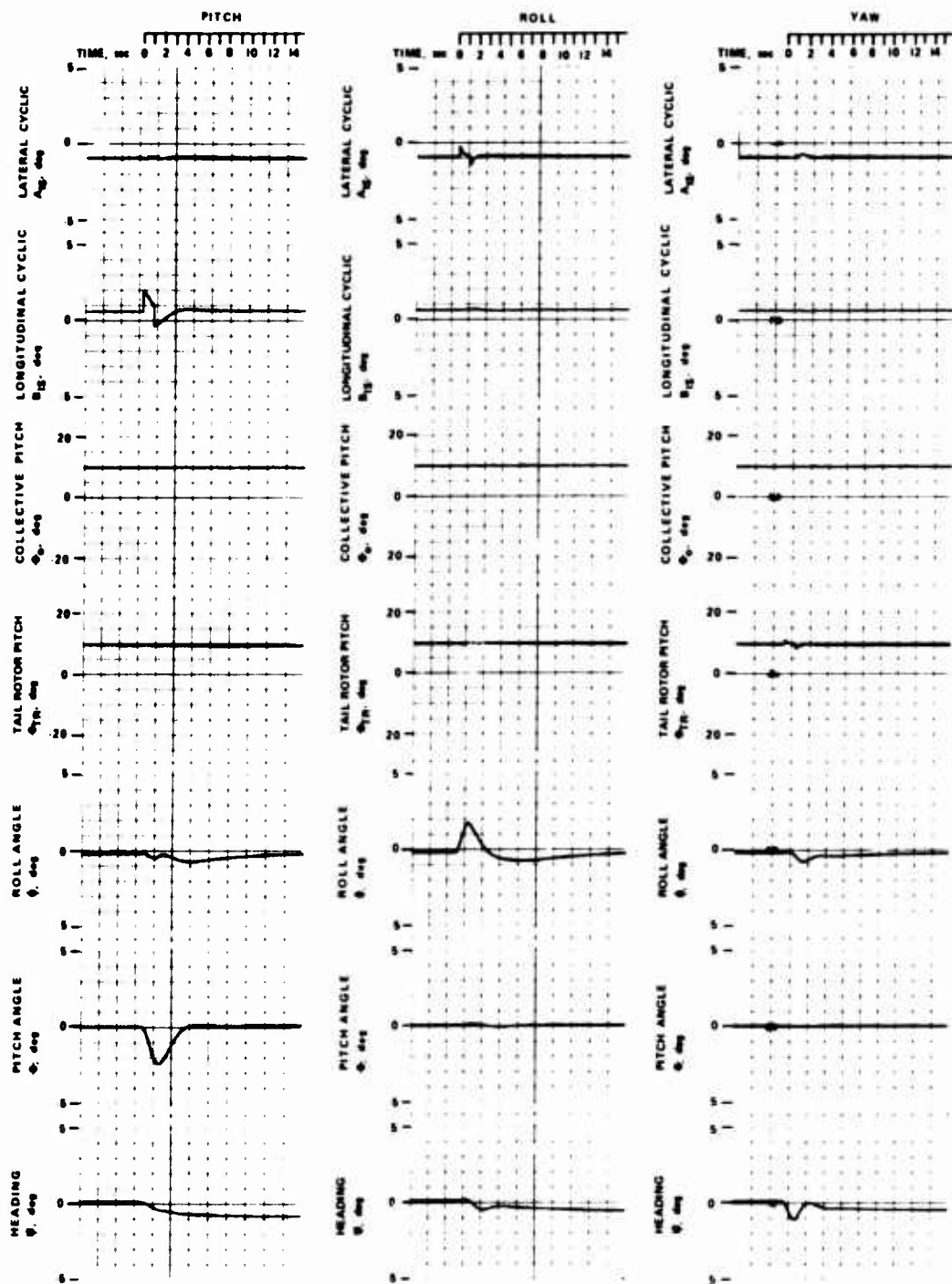


FIGURE 32. AIRCRAFT RESPONSE TO 5% - 1 SECOND CONTROL INPUTS  
FOR GW = 34,000 LB, CG = 328 IN. AT 80 KNOTS.

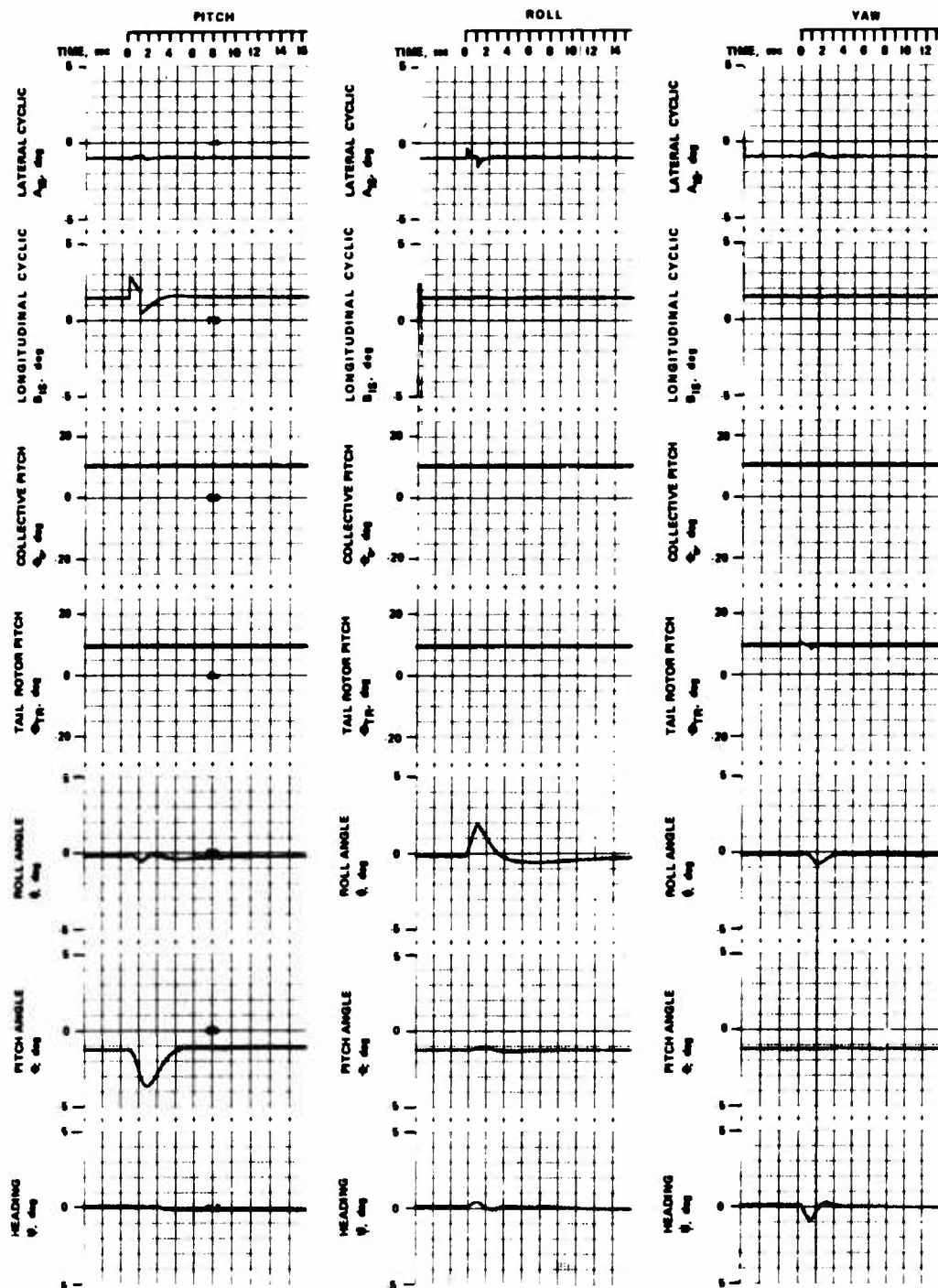


FIGURE 33. AIRCRAFT RESPONSE TO 5% - 1 SECOND CONTROL INPUTS  
FOR GW = 34,000 LB, CG = 328 IN. AT 100 KNOTS.

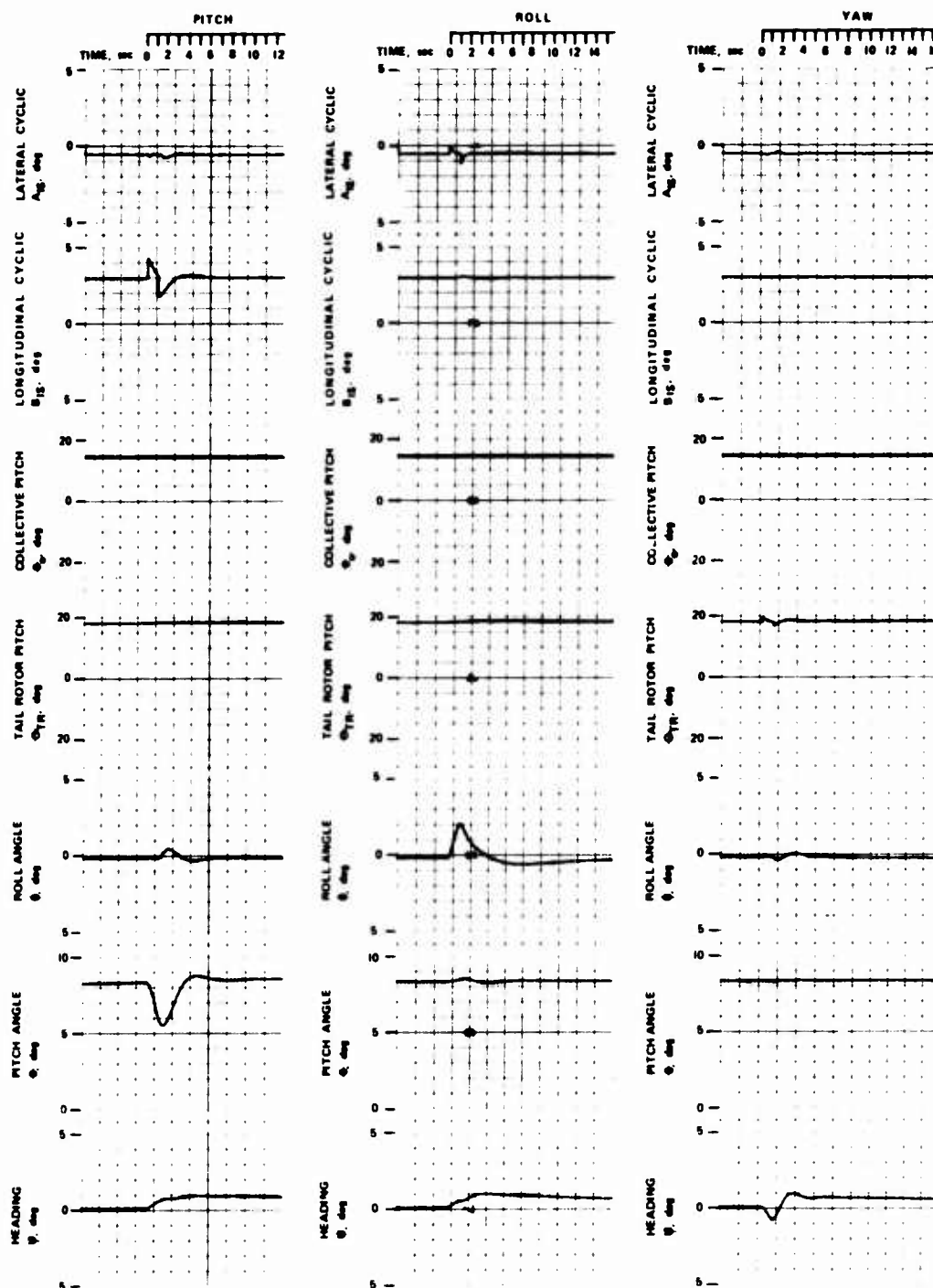


FIGURE 34. AIRCRAFT RESPONSE TO 5% - 1 SECOND CONTROL INPUTS FOR GW = 47,000 LB, CG = 346 IN. AT HOVER.

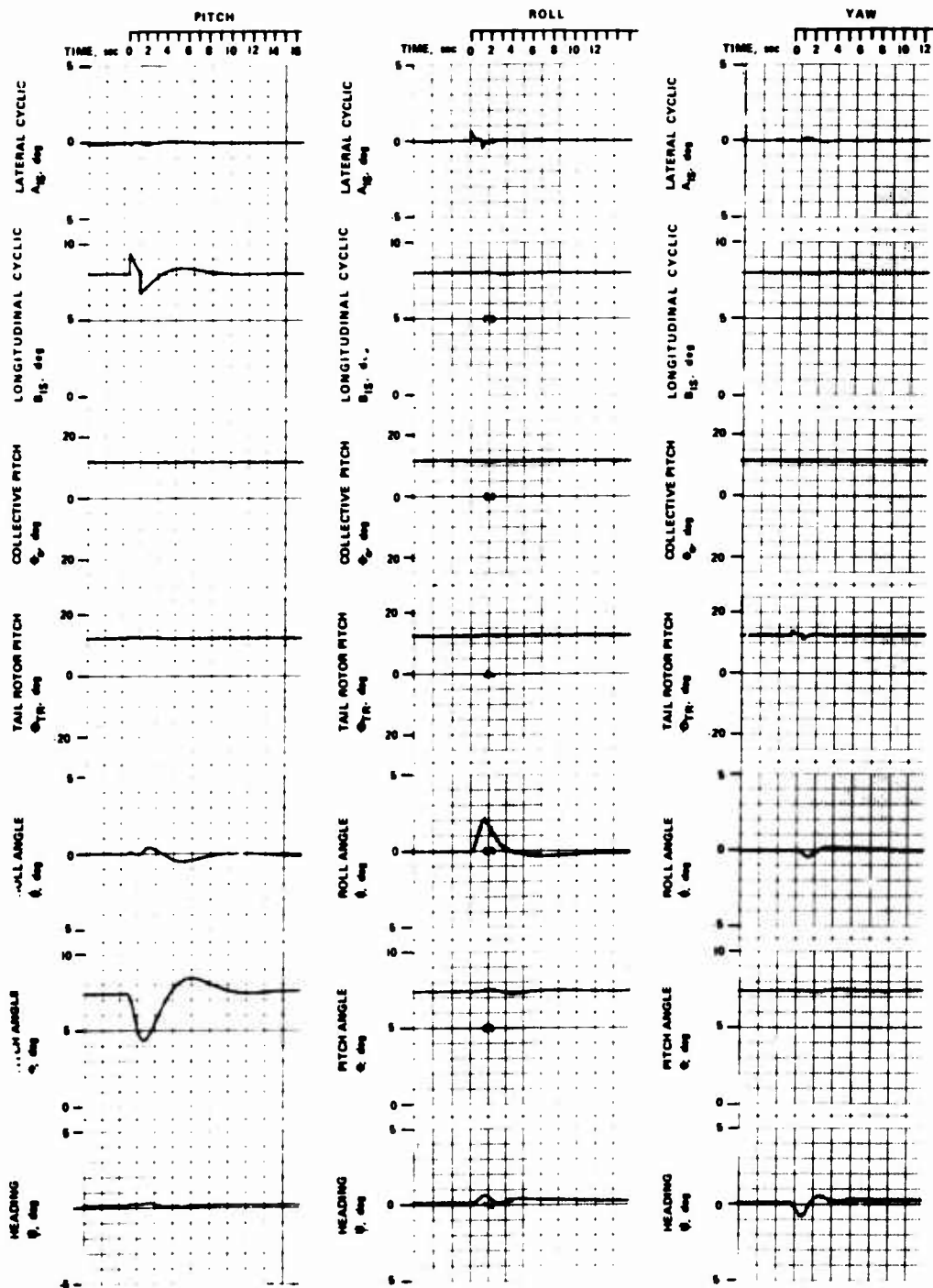


FIGURE 35. AIRCRAFT RESPONSE TO 5% - 1 SECOND CONTROL INPUTS  
FOR GW = 47,000 LB, CG = 348 IN. AT 80 KNOTS.

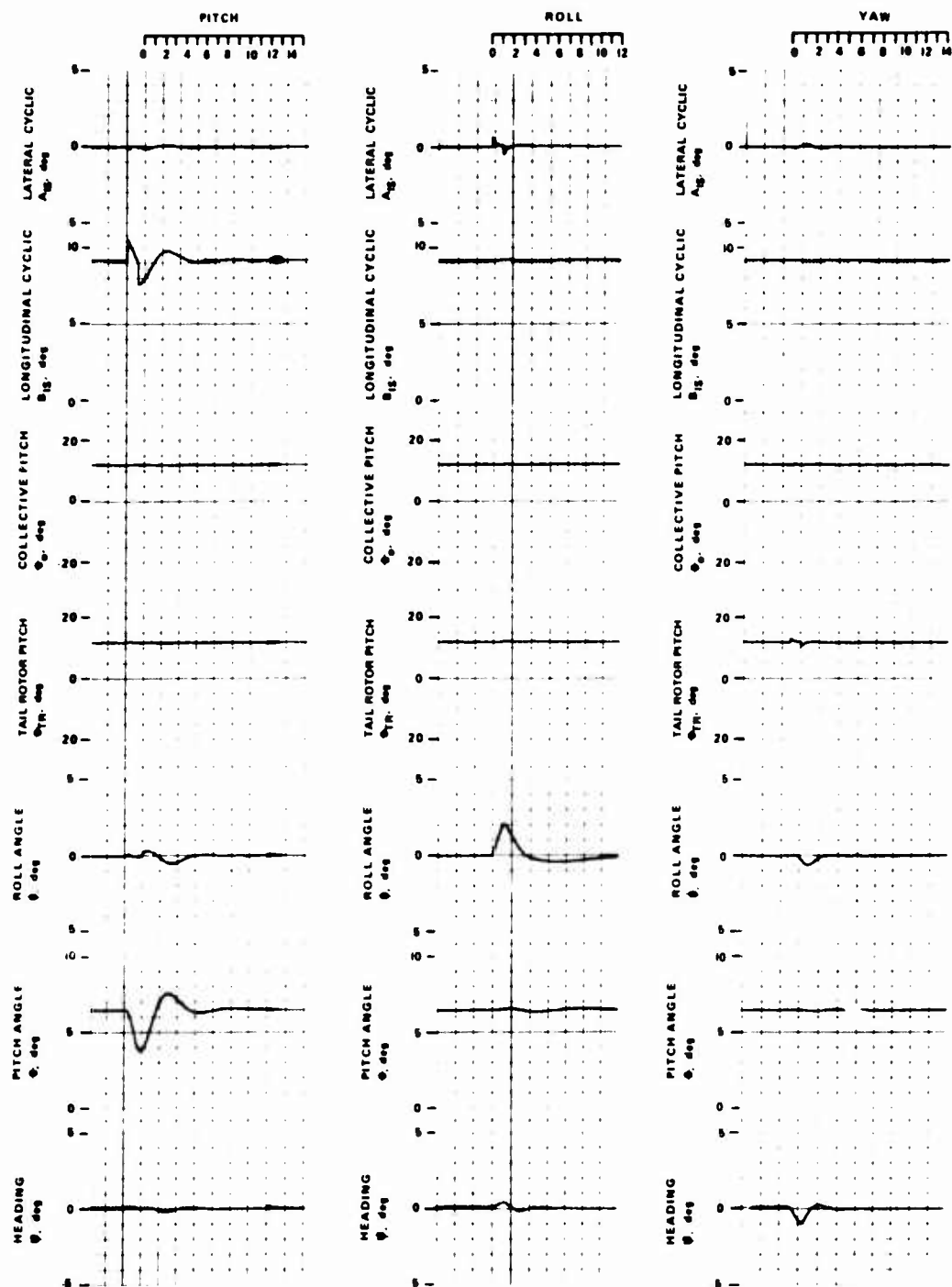


FIGURE 38. AIRCRAFT RESPONSE TO 5% - 1 SECOND CONTROL INPUTS  
FOR GW = 47,000 LB, CG = 348 IN AT 100 KNOTS.

## APPENDIX II

### FLIGHT TEST DATA

This appendix contains the flight recordings taken at various speeds, gross weights and centers of gravity for pulse inputs for the CH-54B helicopter. Maneuvering flight recordings are also included for each loading and speed.

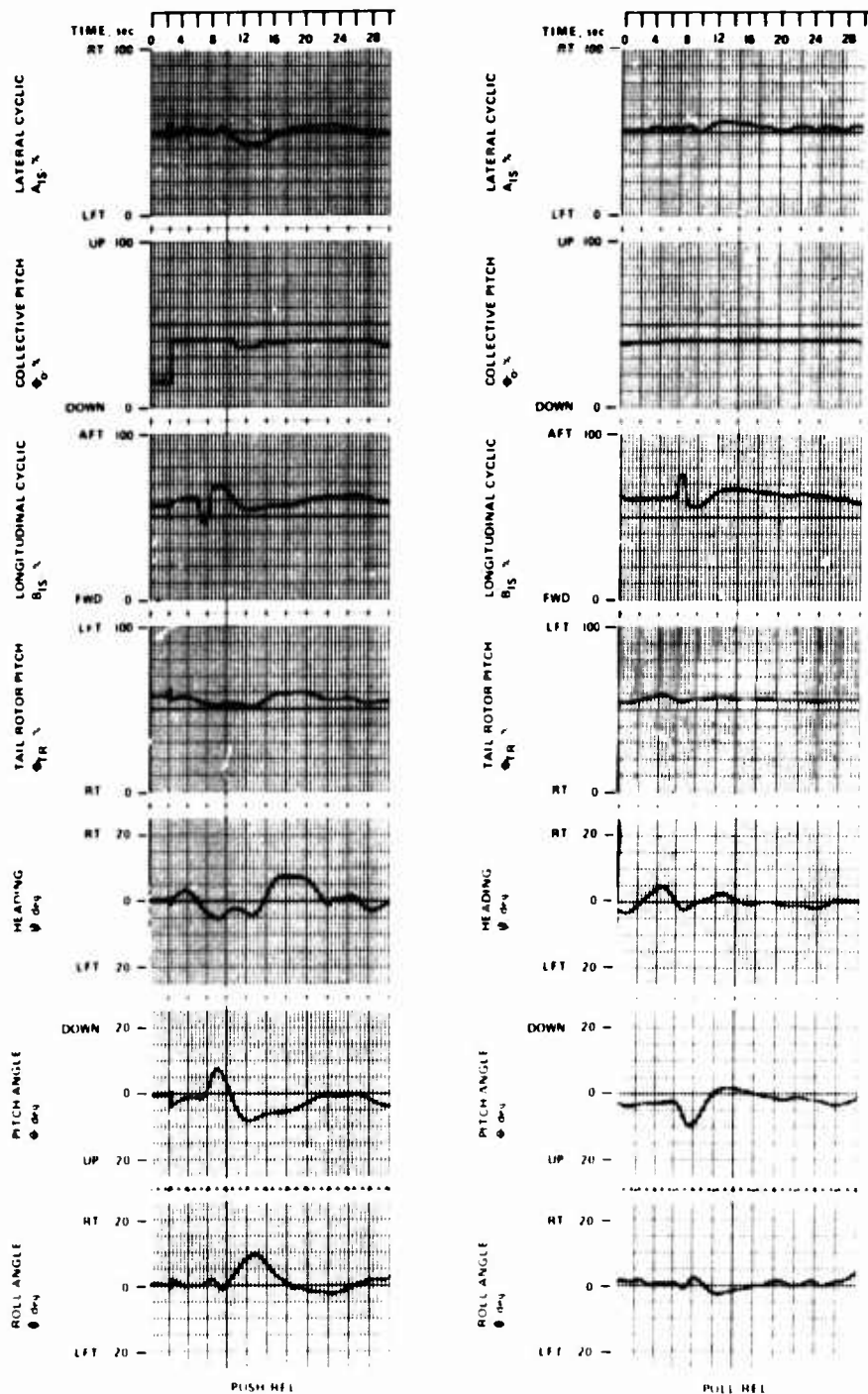
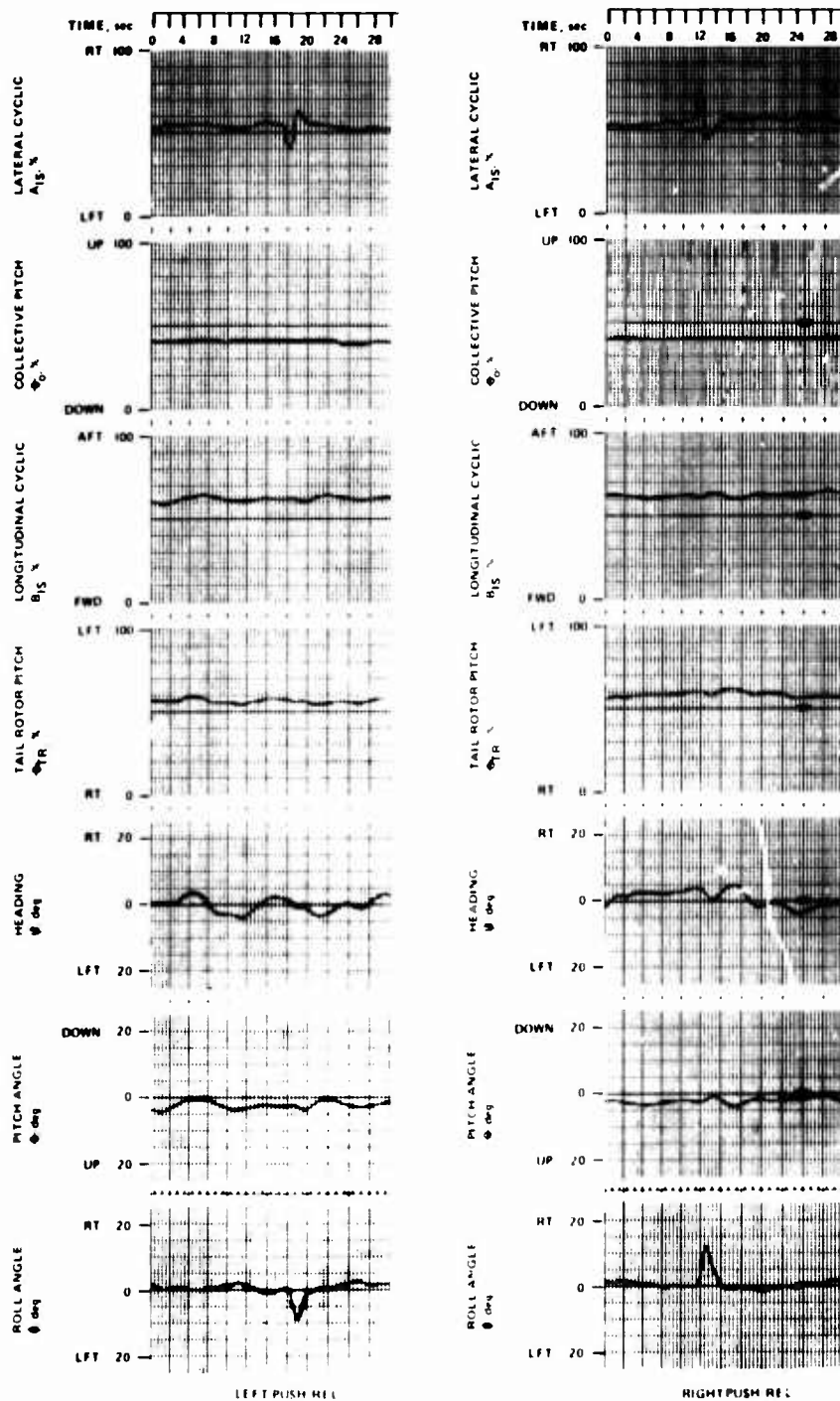
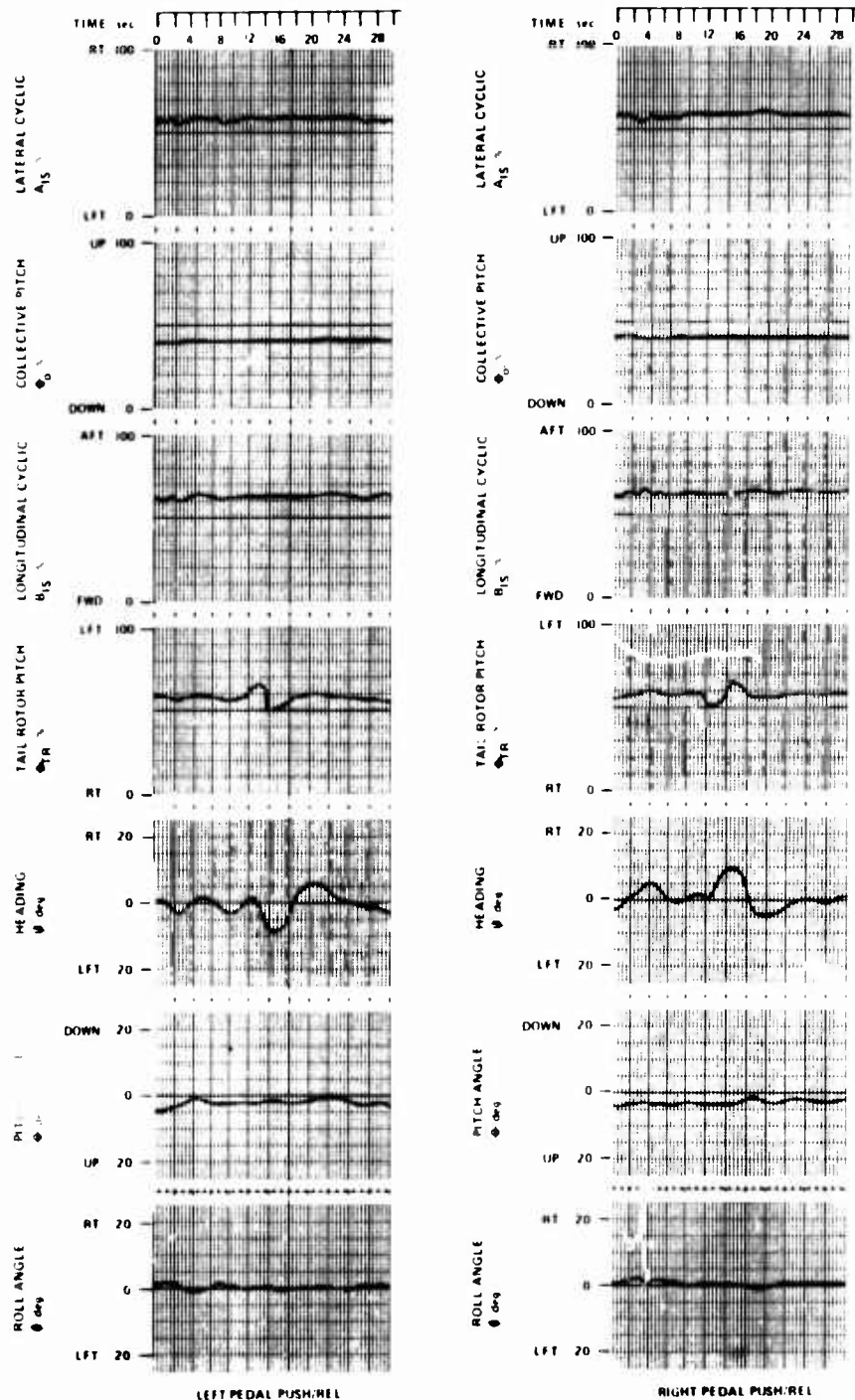


FIGURE 37. AIRCRAFT RESPONSE TO PITCH PULSE INPUTS  
FOR GW = 28,200 LB, CG = 335 IN. AT HOVER.



**FIGURE 38. AIRCRAFT RESPONSE TO ROLL PULSE INPUTS  
FOR GW = 28,200 LB, CG = 335 IN. AT HOVER.**



**FIGURE 39. AIRCRAFT RESPONSE TO YAW PULSE INPUTS  
FOR GW = 28,200 LB, CG = 335 IN. AT HOVER.**

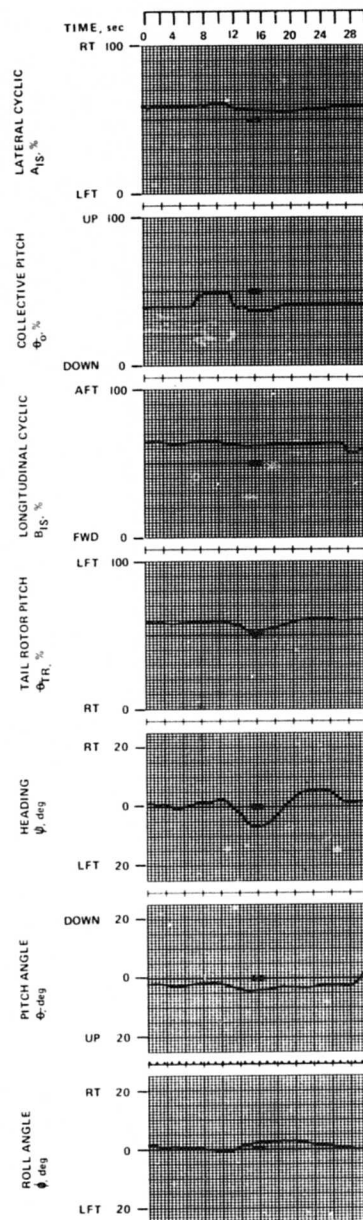
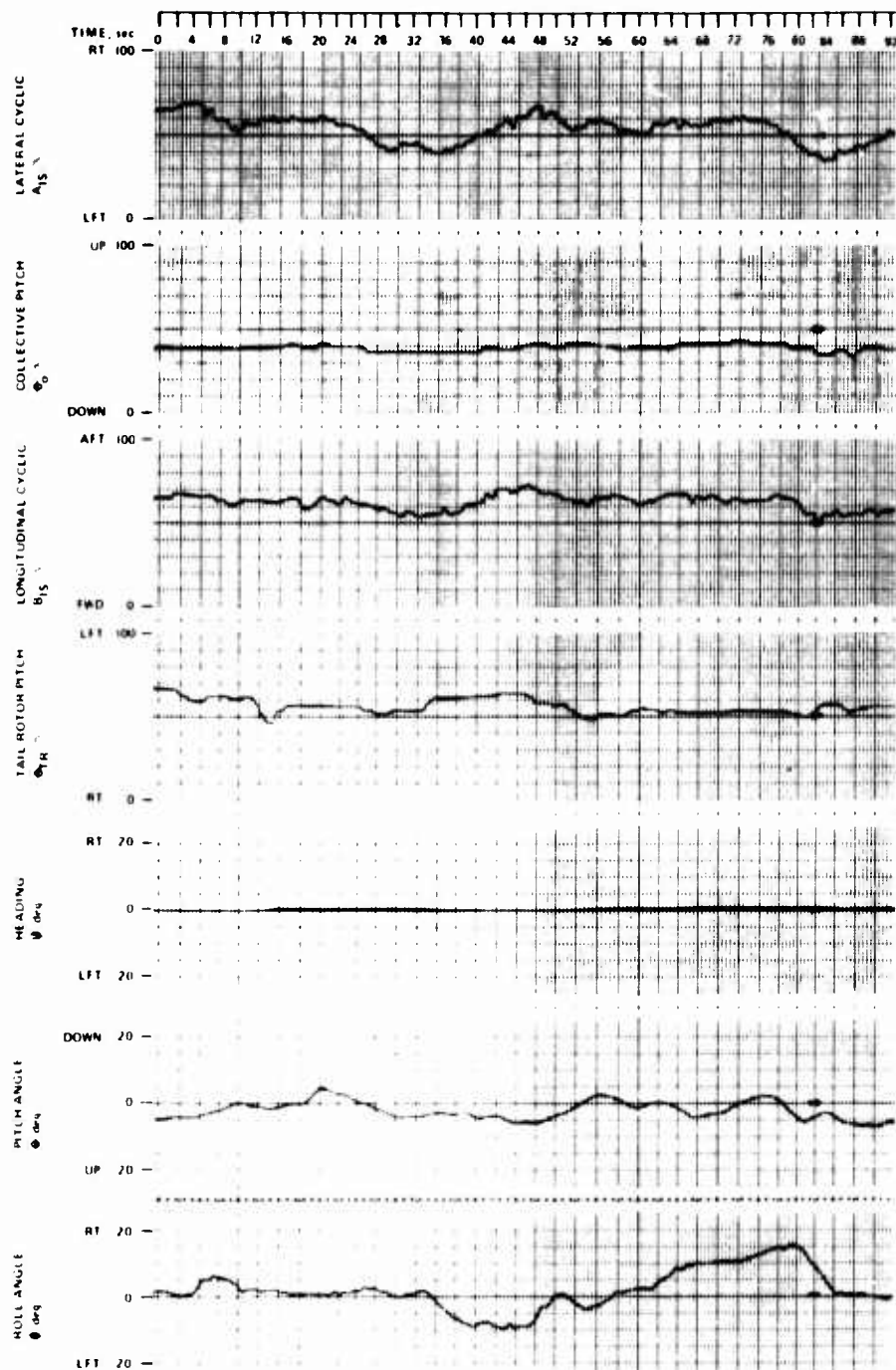


FIGURE 40. AIRCRAFT RESPONSE TO COLLECTIVE STEP INPUTS  
FOR GW = 28,200 LB, CG = 335 IN. AT HOVER.



**FIGURE 4I. HOVER MANEUVERING FLIGHT - SLOW FLIGHT AND S TURNS  
FOR GW = 28,200 LB, CG = 335 IN. AT HOVER.**

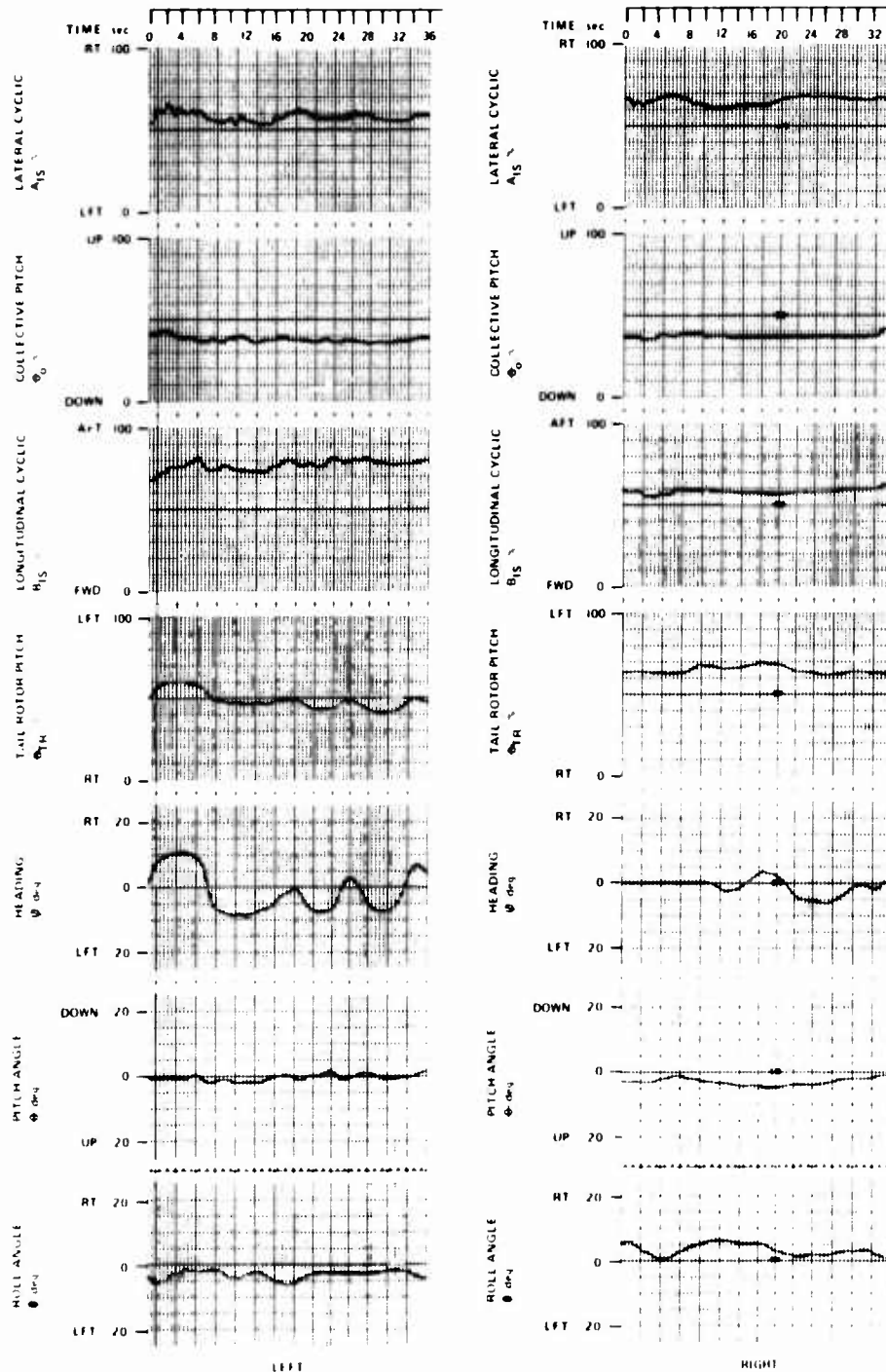
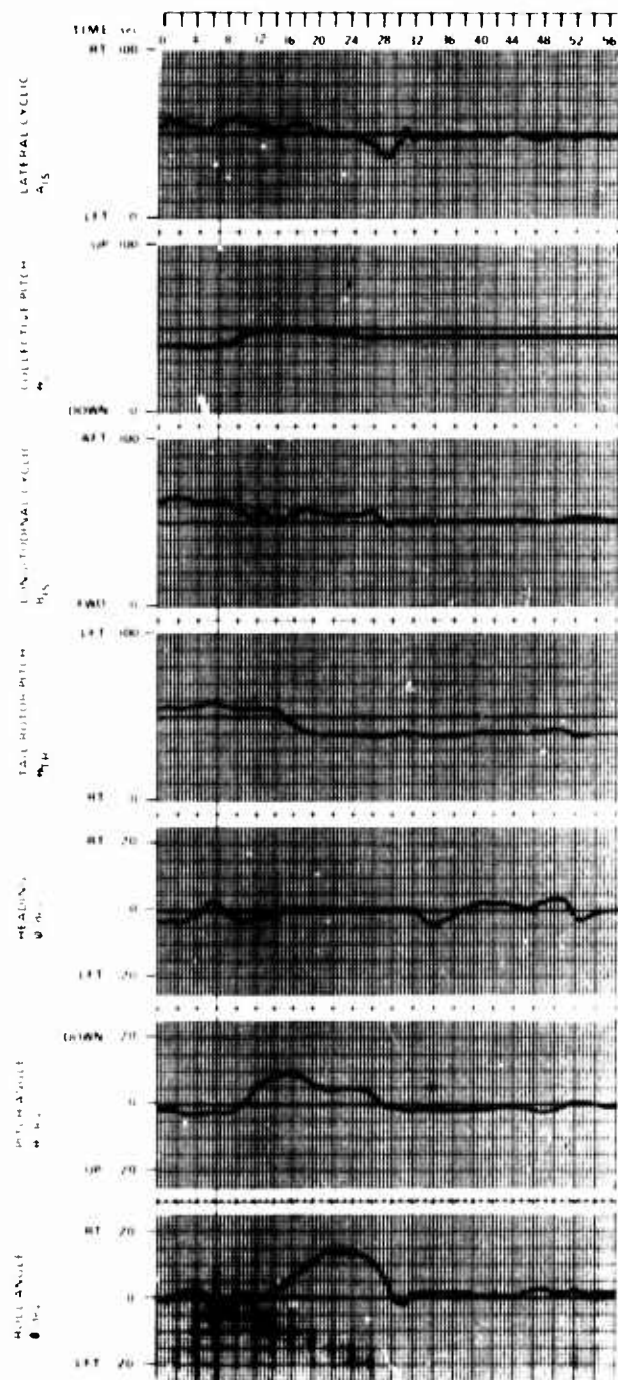
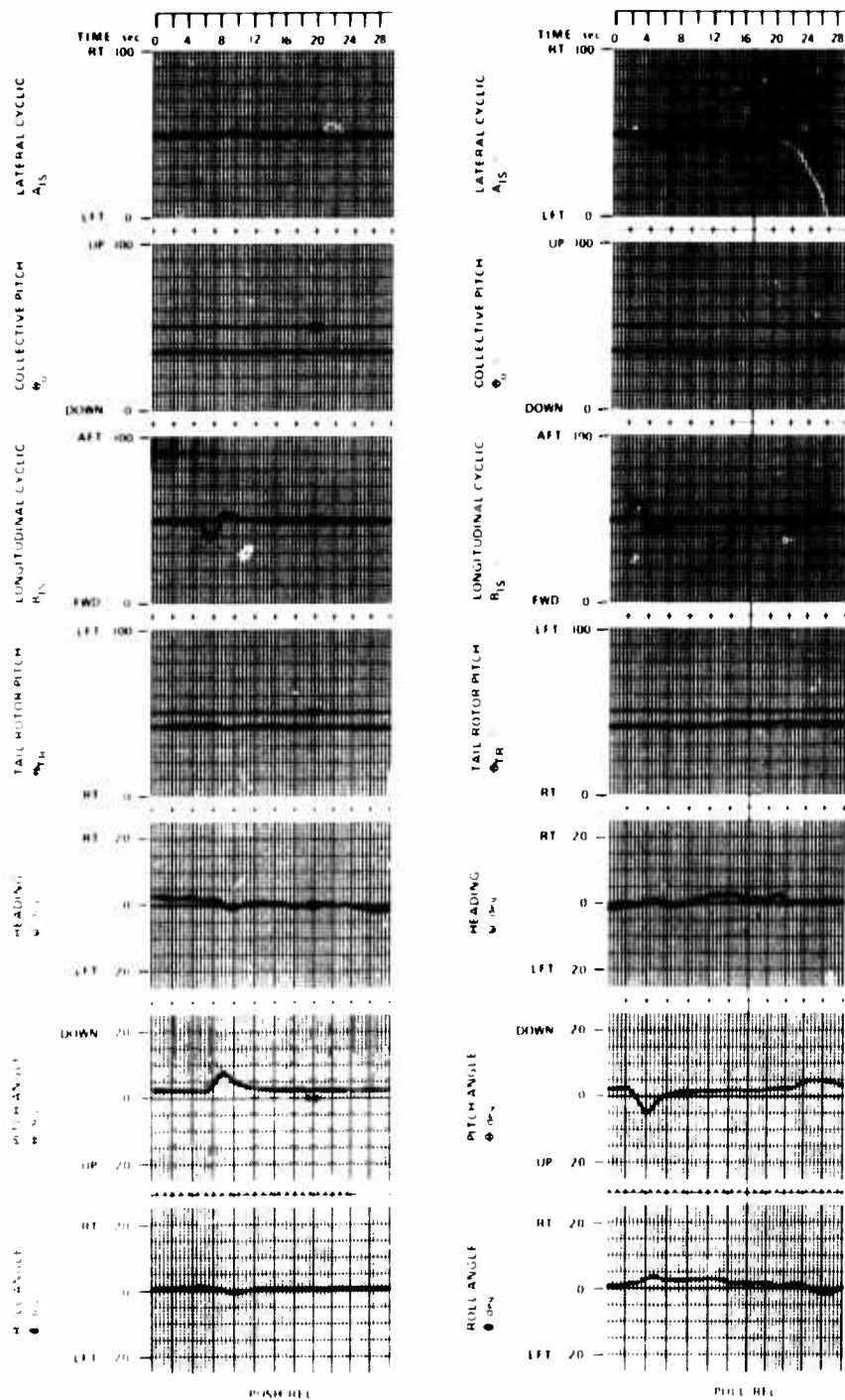


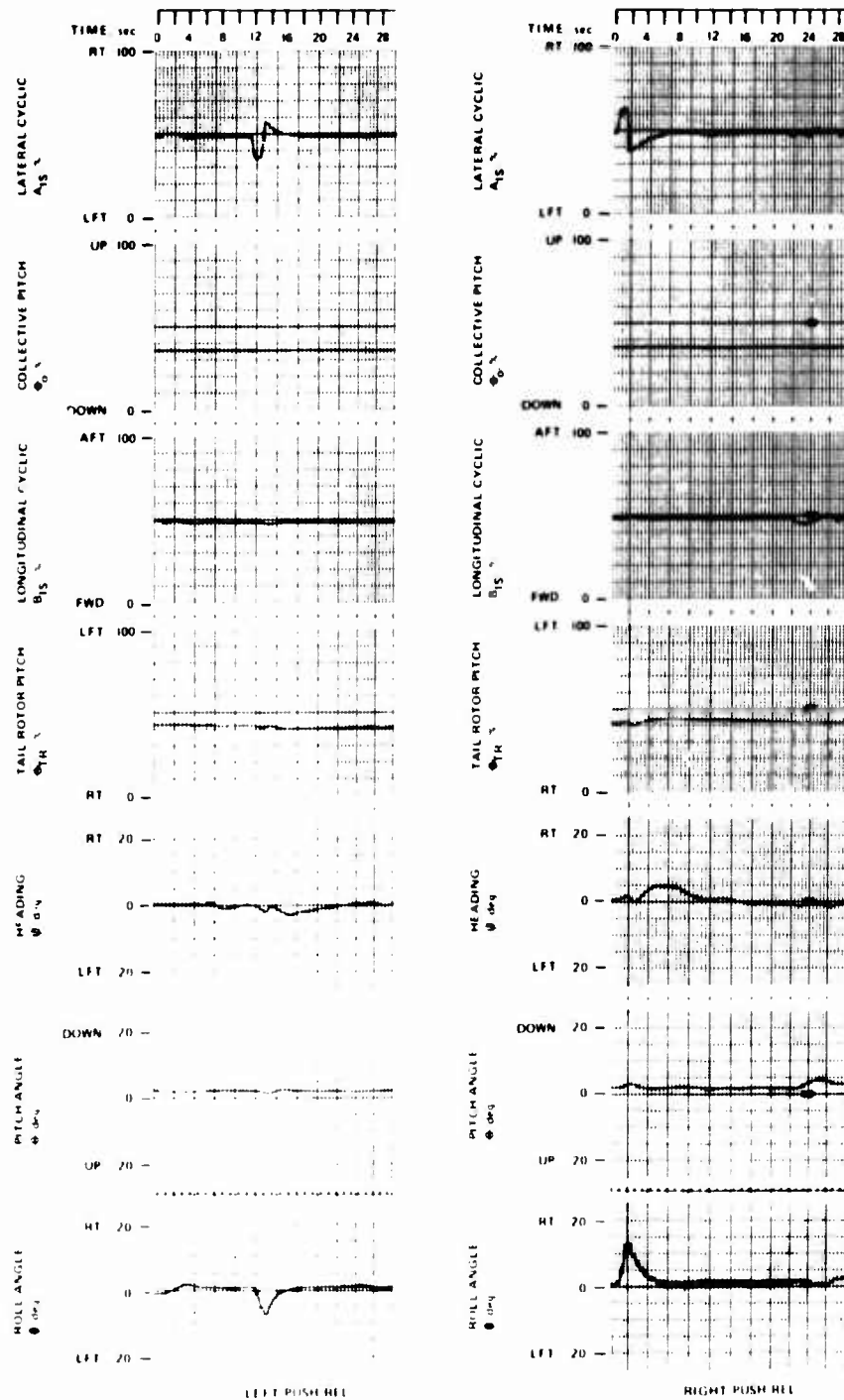
FIGURE 42. AIRCRAFT RESPONSE TO SIDEWARD FLIGHT  
FOR GW = 28,200 LB, CG = 335 IN. AT HOVER.



**FIGURE 43. AIRCRAFT RESPONSE TO TAKE OFF  
FOR GW = 28,000 LB, CG = 335 IN. AT HOVER**



**FIGURE 44. AIRCRAFT RESPONSE TO PITCH PULSE INPUTS FOR GW = 28,200 LB, CG = 335 IN. AT 80 KNOTS.**



**FIGURE 45. AIRCRAFT RESPONSE TO ROLL PULSE INPUTS  
FOR GW = 28,200 LB, CG = 335 IN. AT 80 KNOTS.**

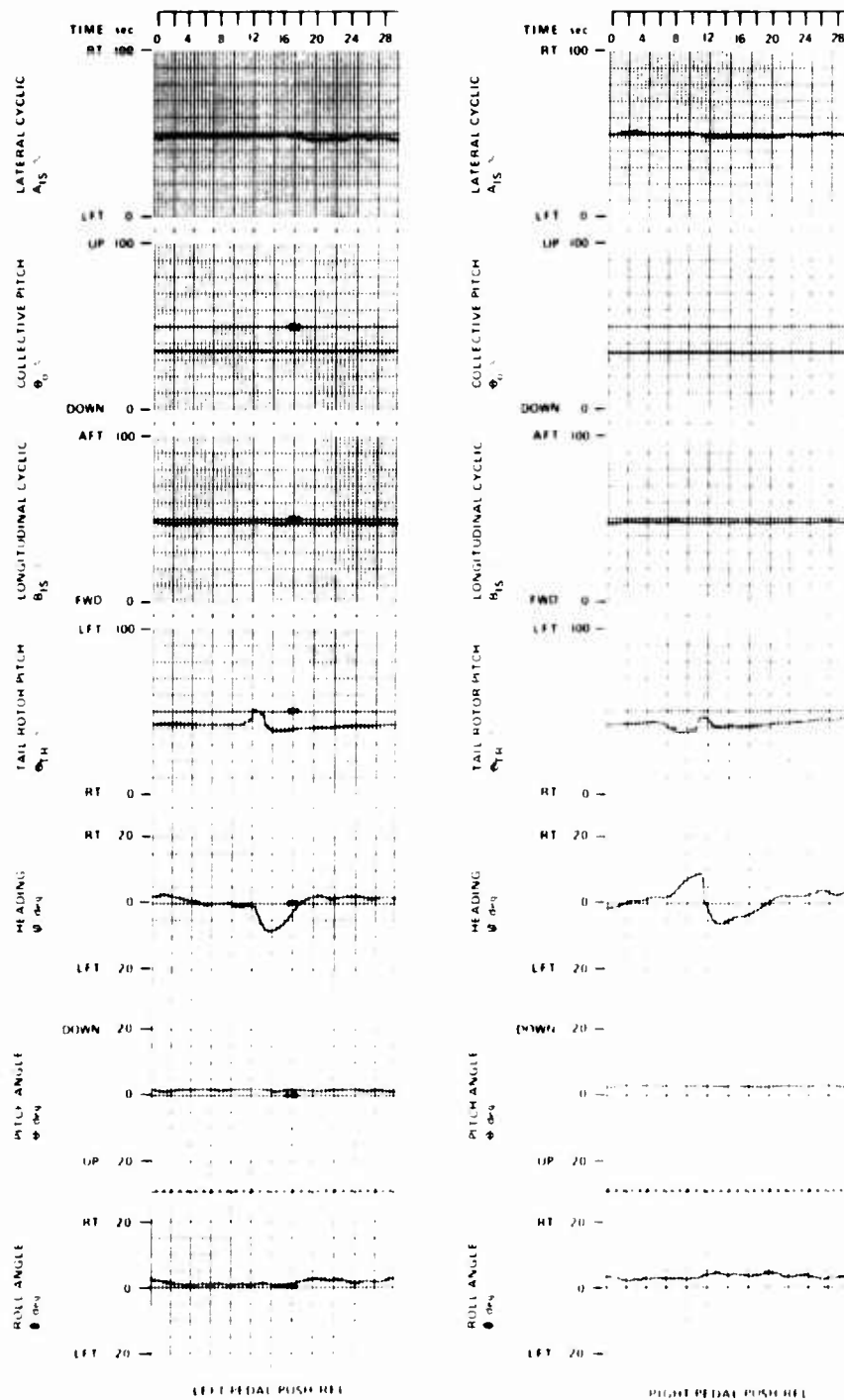
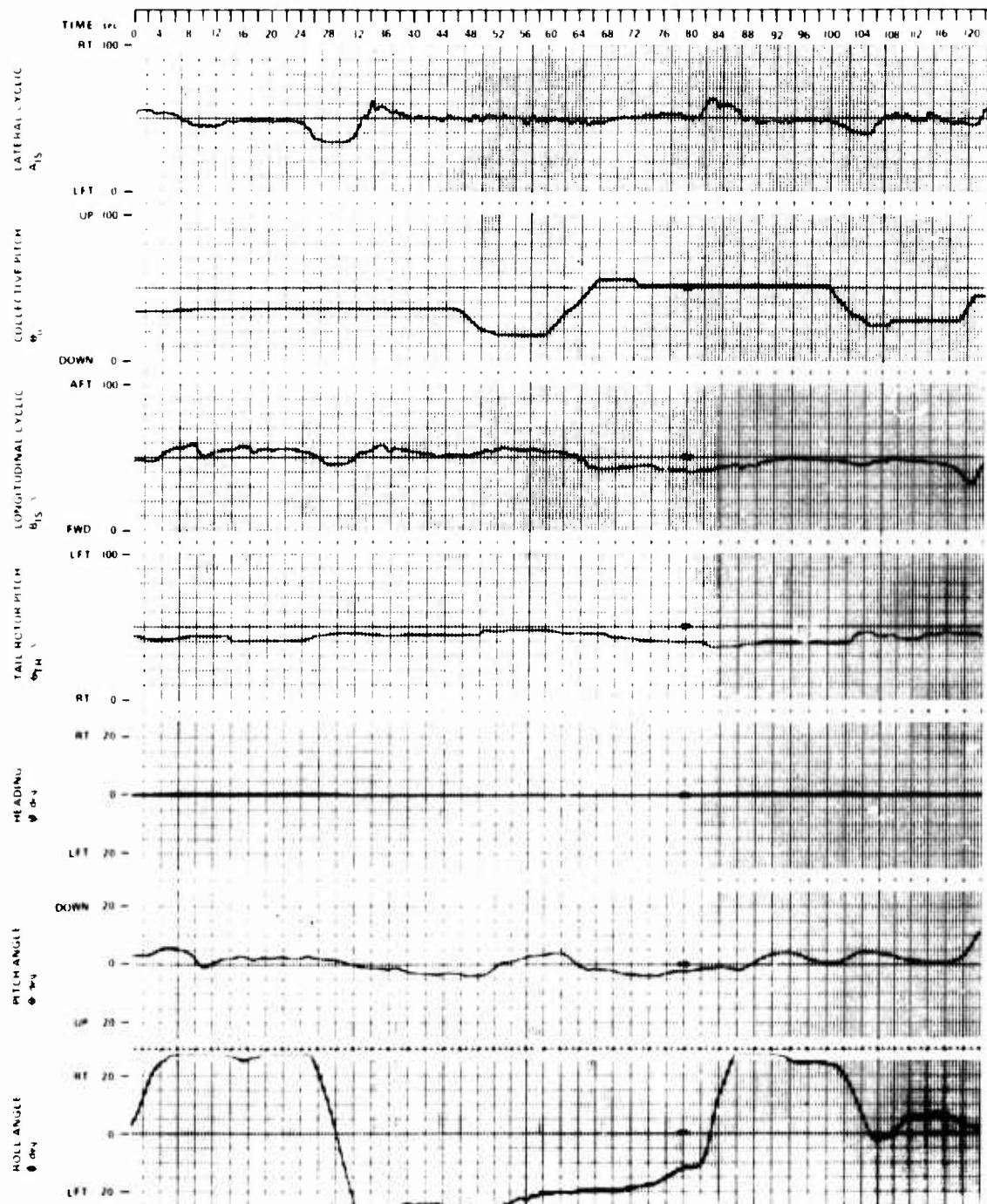
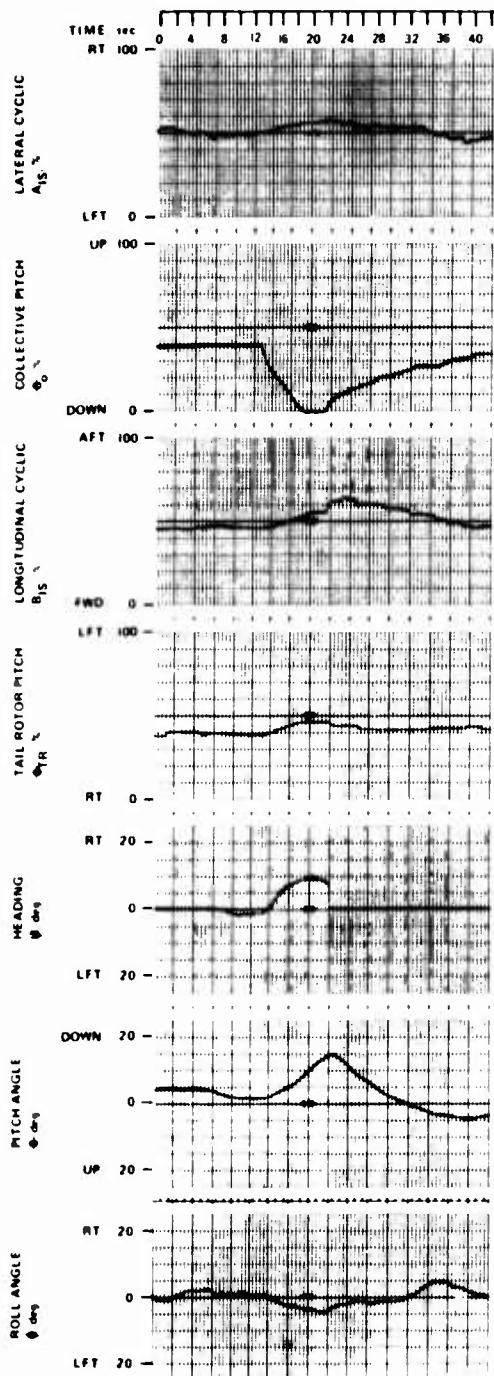


FIGURE 46. AIRCRAFT RESPONSE TO YAW PULSE INPUTS  
FOR GW = 28,200 LB, CG = 335 IN. AT 80 KNOTS.



**FIGURE 47. FORWARD FLIGHT MANEUVERING - CLIMBING AND DESCENDING  
LEFT AND RIGHT TURNS FOR GW = 28,200 LB, CG = 335 IN. AT 80 KNOTS.**



**FIGURE 48. ENTRY TO AUTOROTATION  
FOR GW = 28,200 LB, CG = 335 IN. AT 80 KNOTS.**

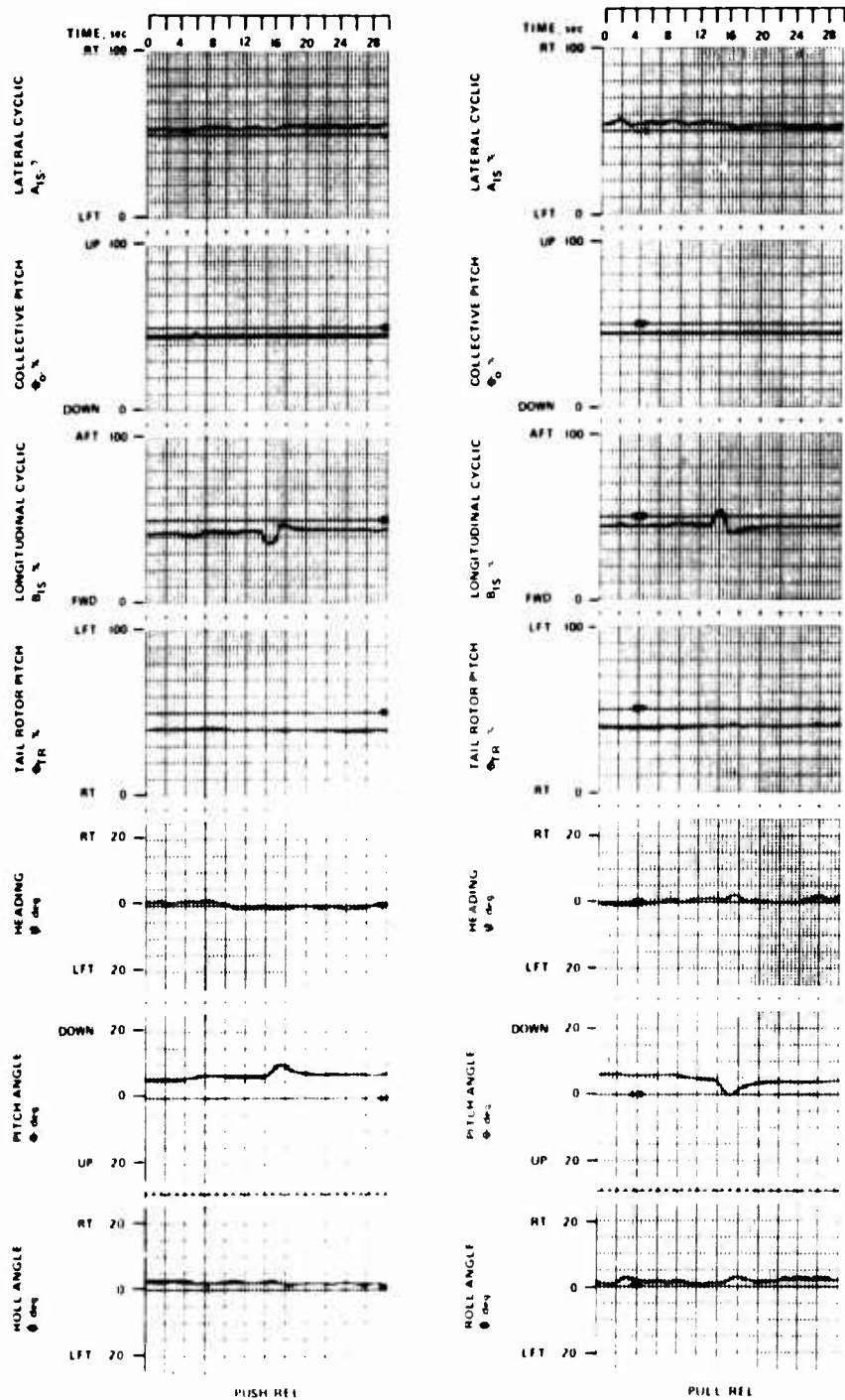
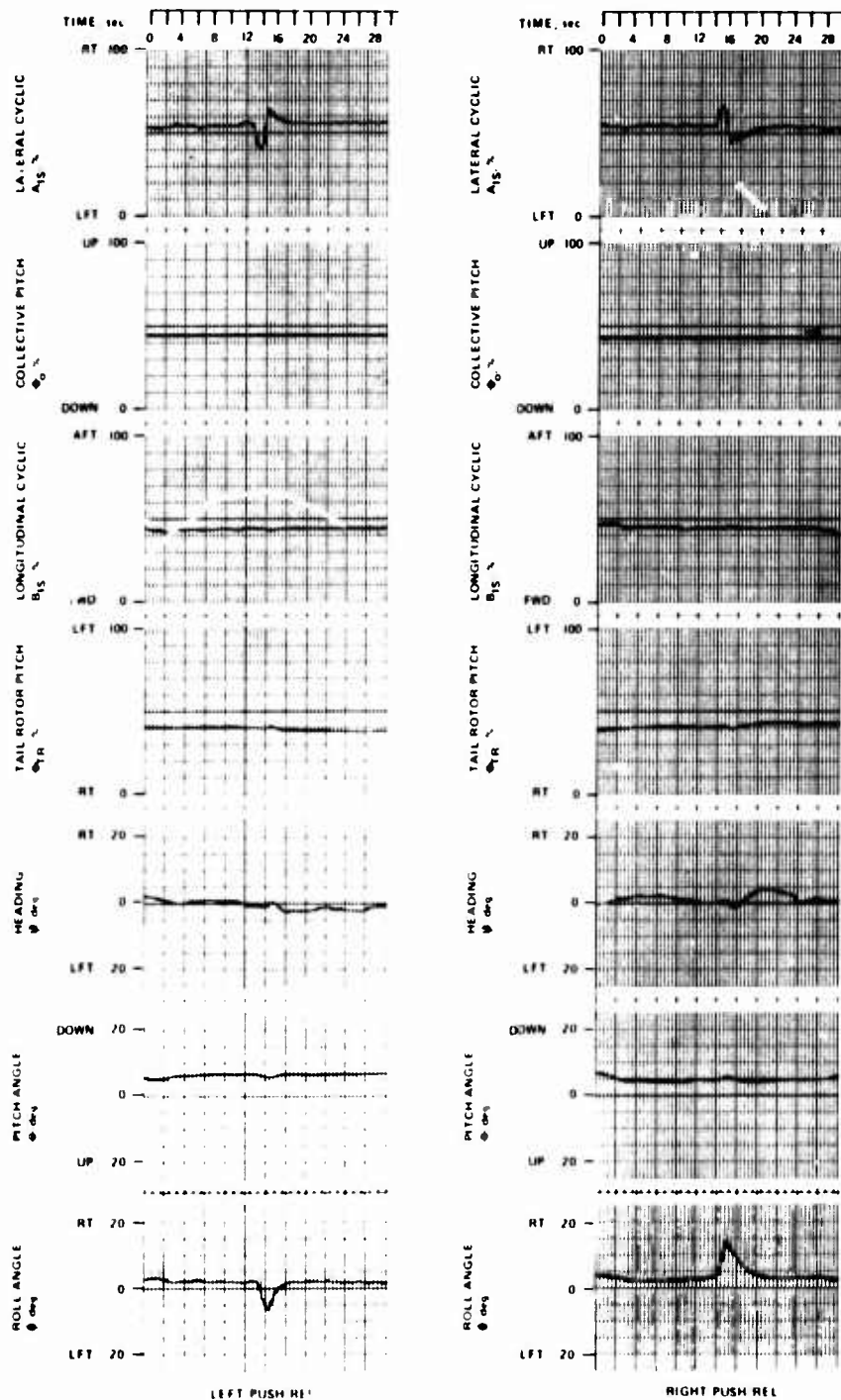


FIGURE 49. AIRCRAFT RESPONSE TO PITCH PULSE INPUTS  
FOR GW = 28,200 LB, CG = 335 IN. AT  $V_{MAX}$ .



**FIGURE 50. AIRCRAFT RESPONSE TO ROLL PULSE INPUTS  
FOR GW = 28,200 LB, CG = 335 IN. AT  $V_{MAX}$ .**

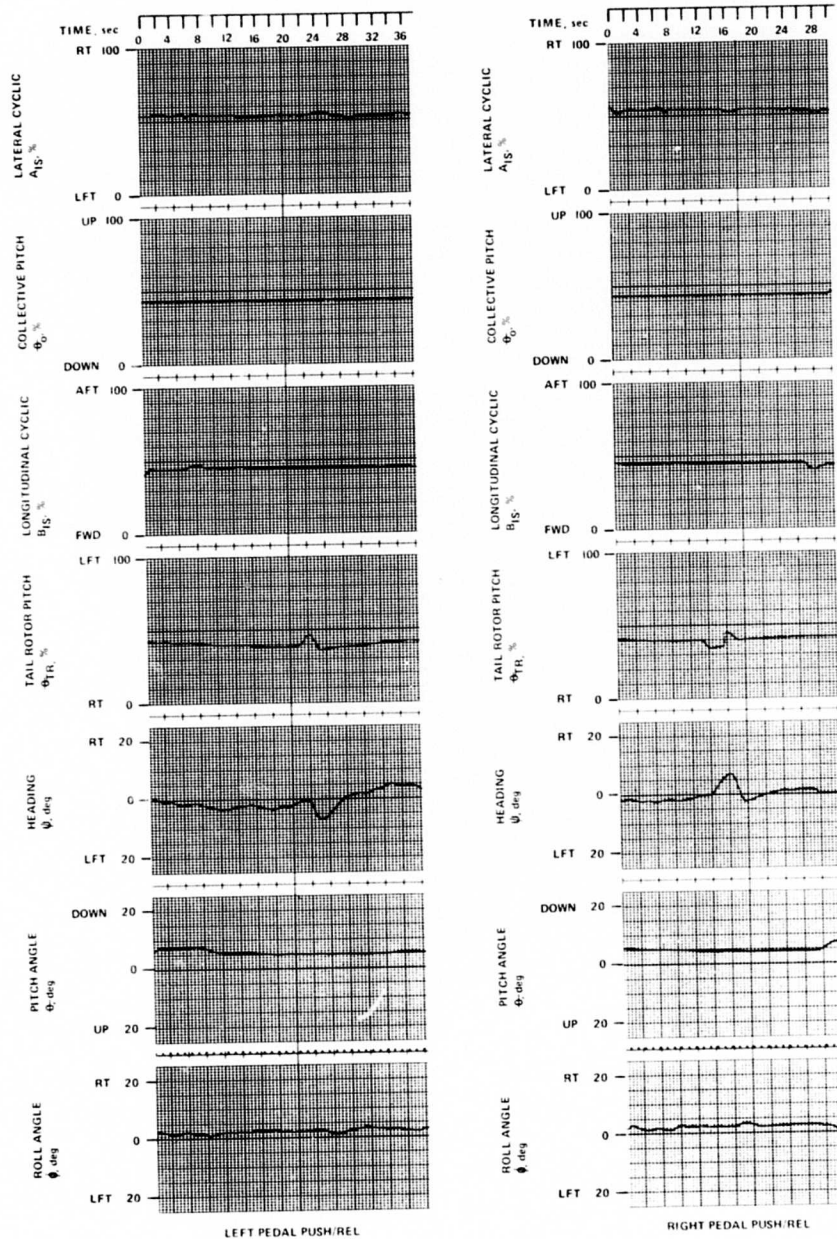


FIGURE 51. AIRCRAFT RESPONSE TO YAW PULSE INPUTS  
FOR GW = 28,200 LB, CG = 335 IN. AT  $V_{MAX}$ .

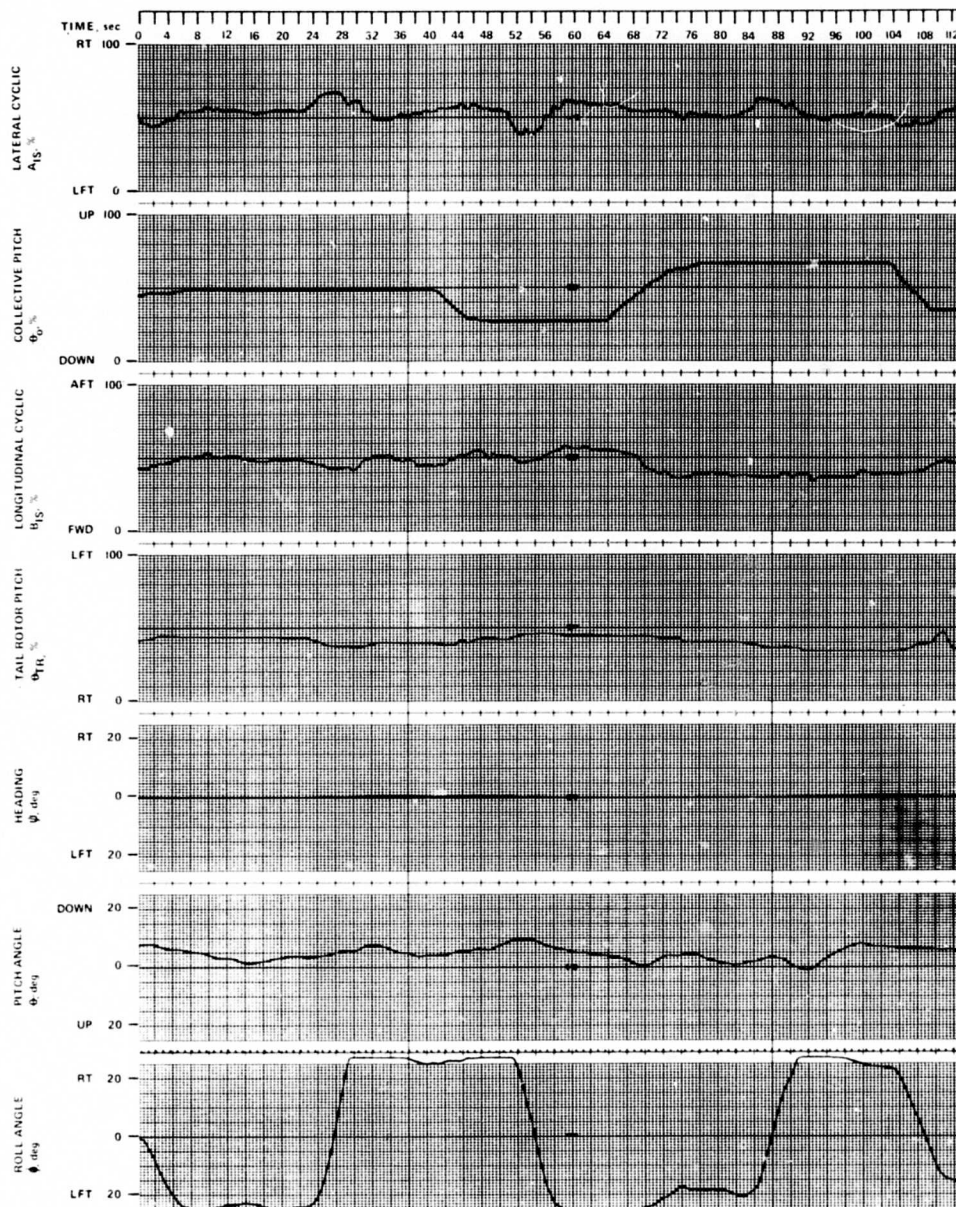


FIGURE 52. FORWARD FLIGHT MANEUVERING - CLIMBING AND DESCENDING  
LEFT AND RIGHT TURNS FOR GW = 28,200 LB, CG = 335 IN. AT  $V_{MAX}$ .

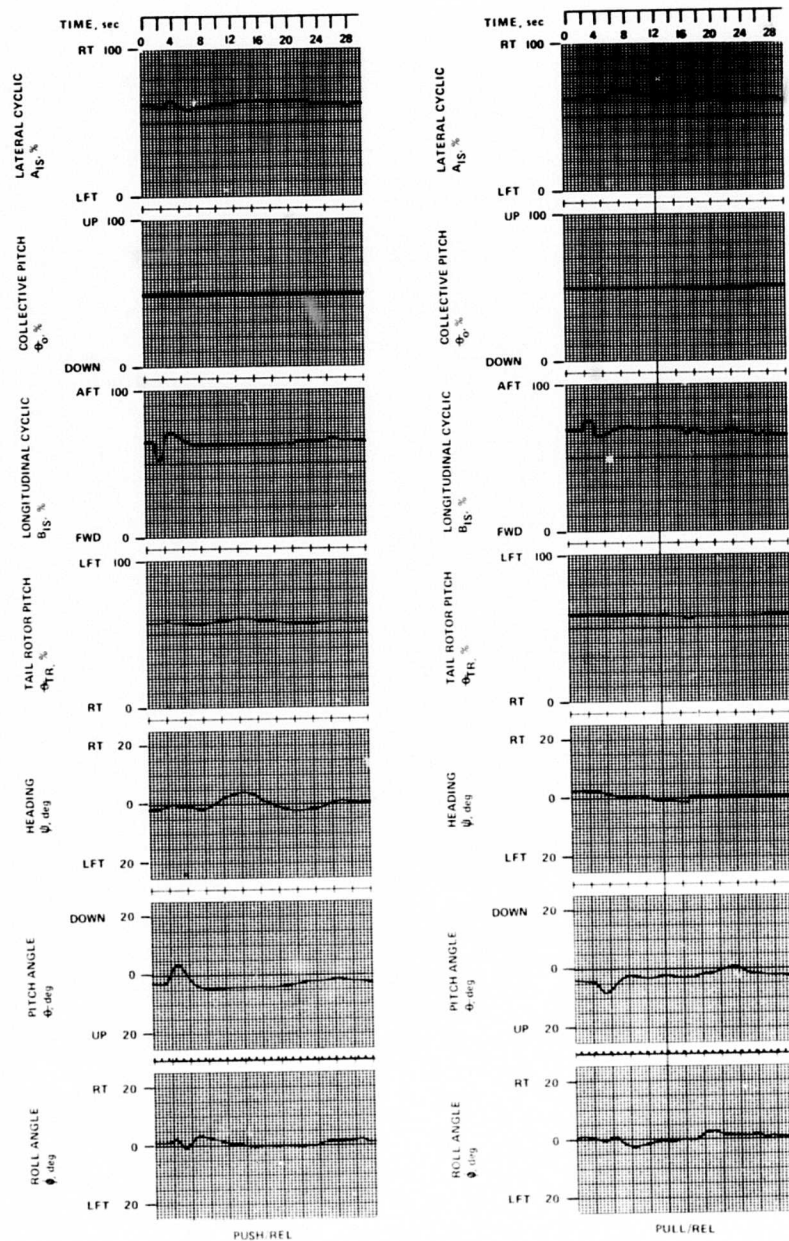


FIGURE 53. AIRCRAFT RESPONSE TO PITCH PULSE INPUTS  
FOR GW = 35,000 LB, CG = 336 IN. AT HOVER.

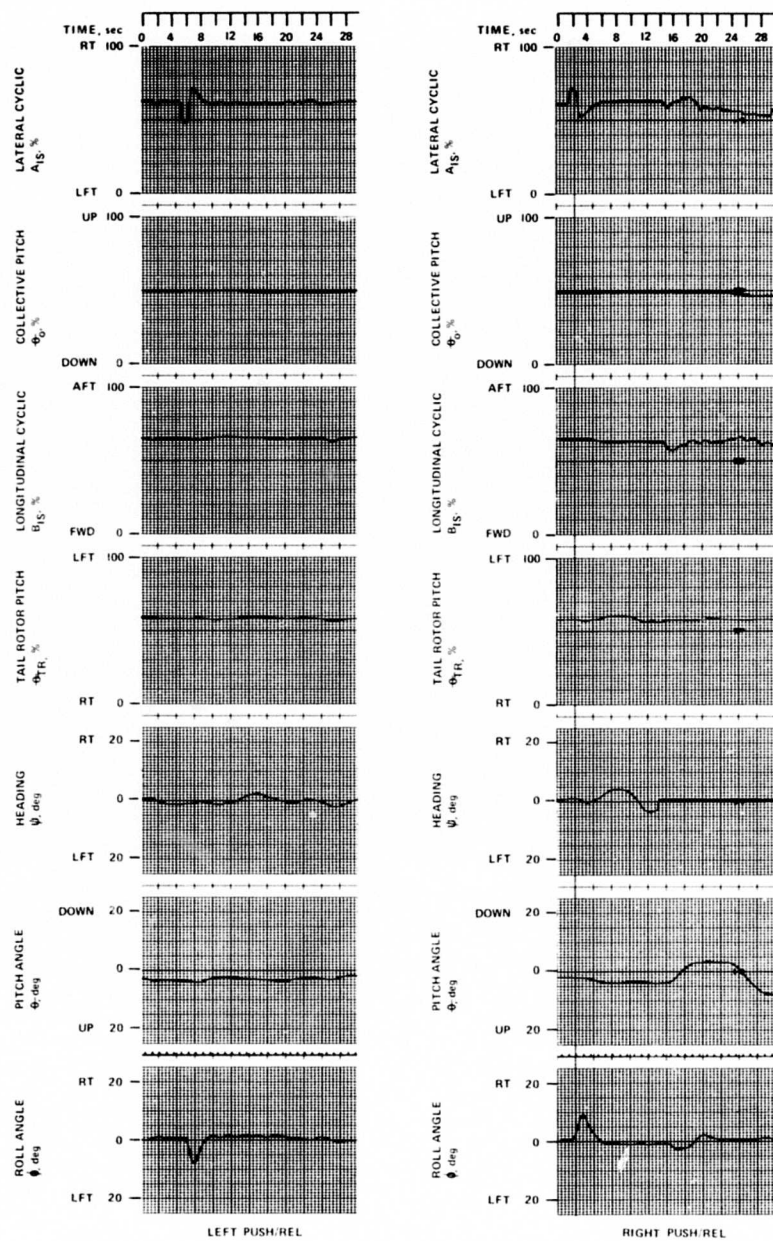
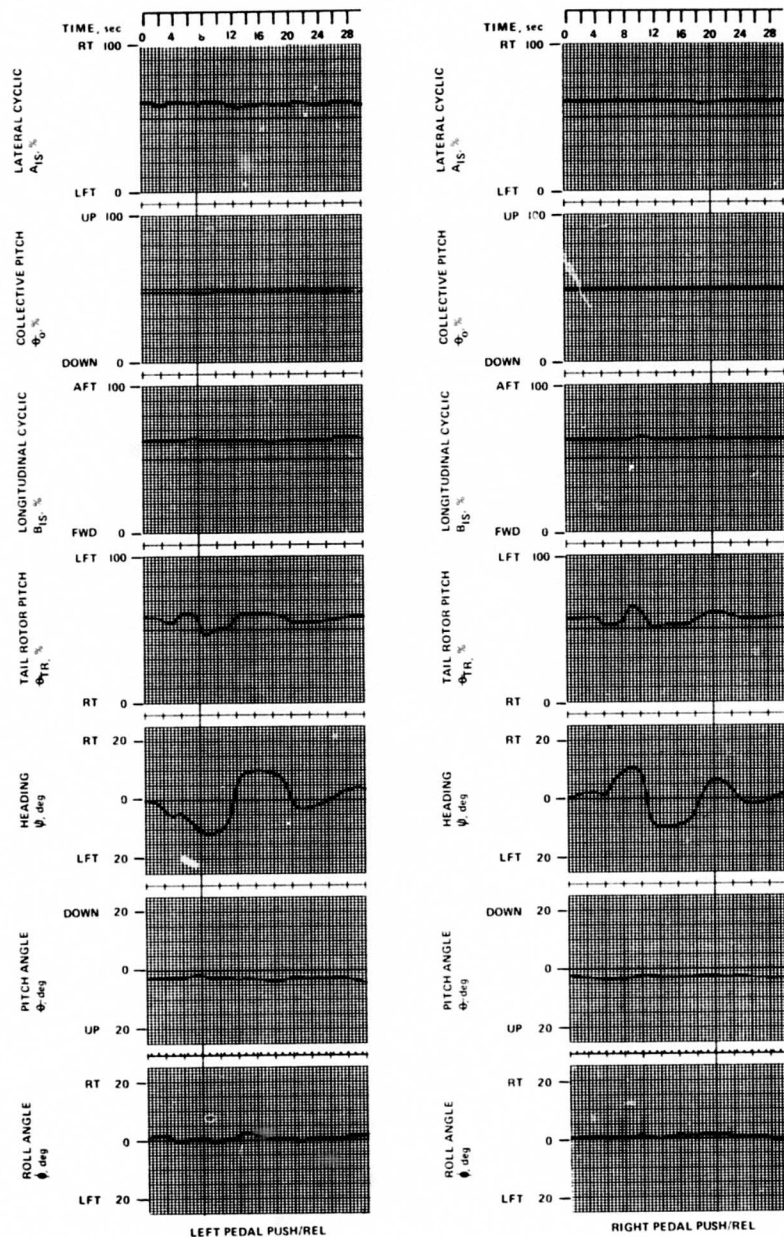


FIGURE 54. AIRCRAFT RESPONSE TO ROLL PULSE INPUTS  
FOR GW = 35,000 LB, CG = 336 IN. AT HOVER.



**FIGURE 55. AIRCRAFT RESPONSE TO YAW PULSE INPUTS  
FOR GW = 35,000 LB, CG = 336 IN. AT HOVER.**

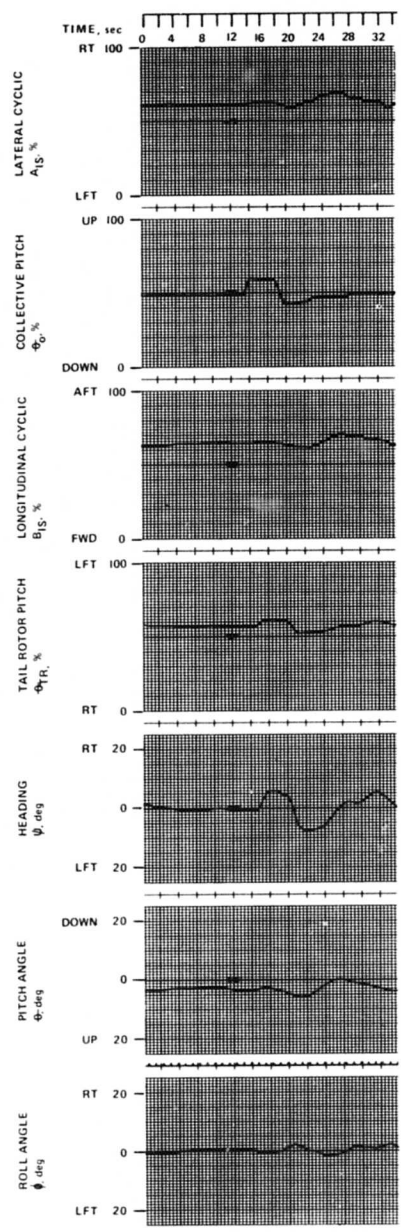


FIGURE 56. AIRCRAFT RESPONSE TO COLLECTIVE STEP INPUTS  
FOR GW = 35,000 LB, CG = 336 IN. AT HOVER.

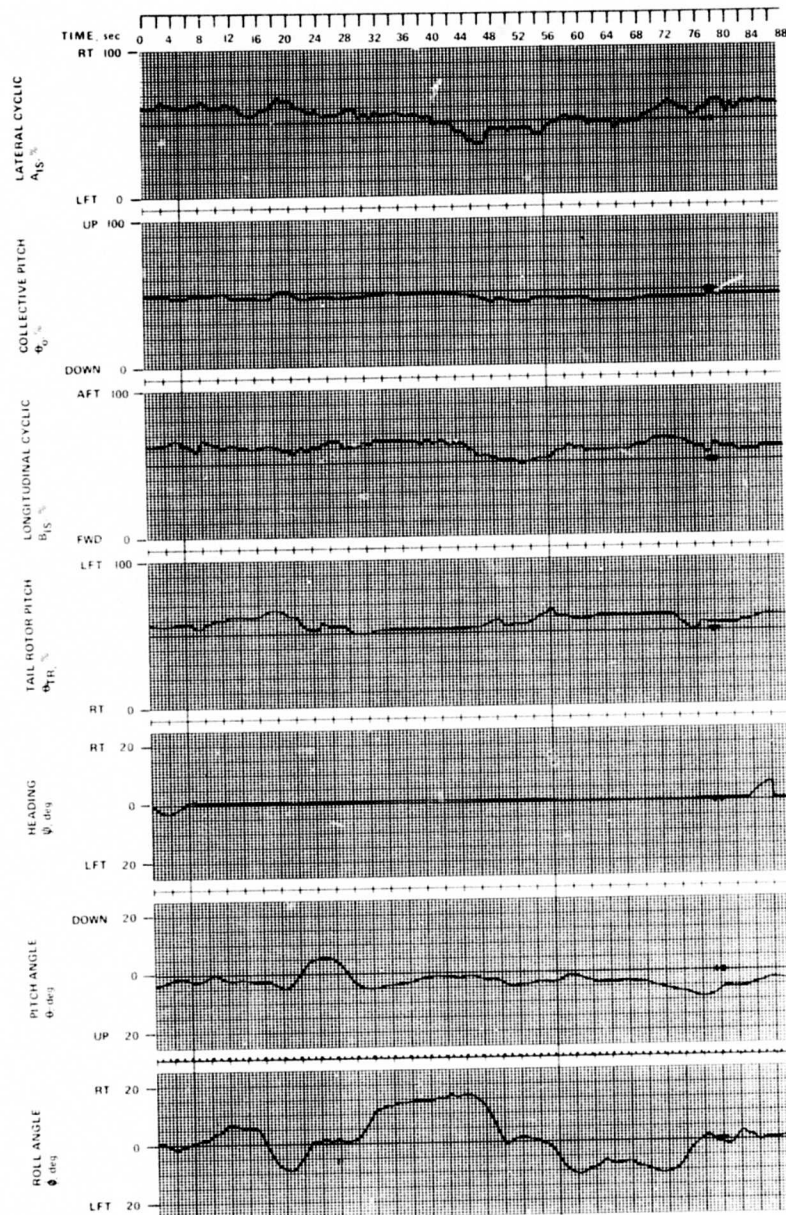


FIGURE 57. HOVER MANEUVERING FLIGHT - SLOW FLIGHT AND S TURNS  
FOR GW = 35,000 LB, CG = 336 IN. AT HOVER.

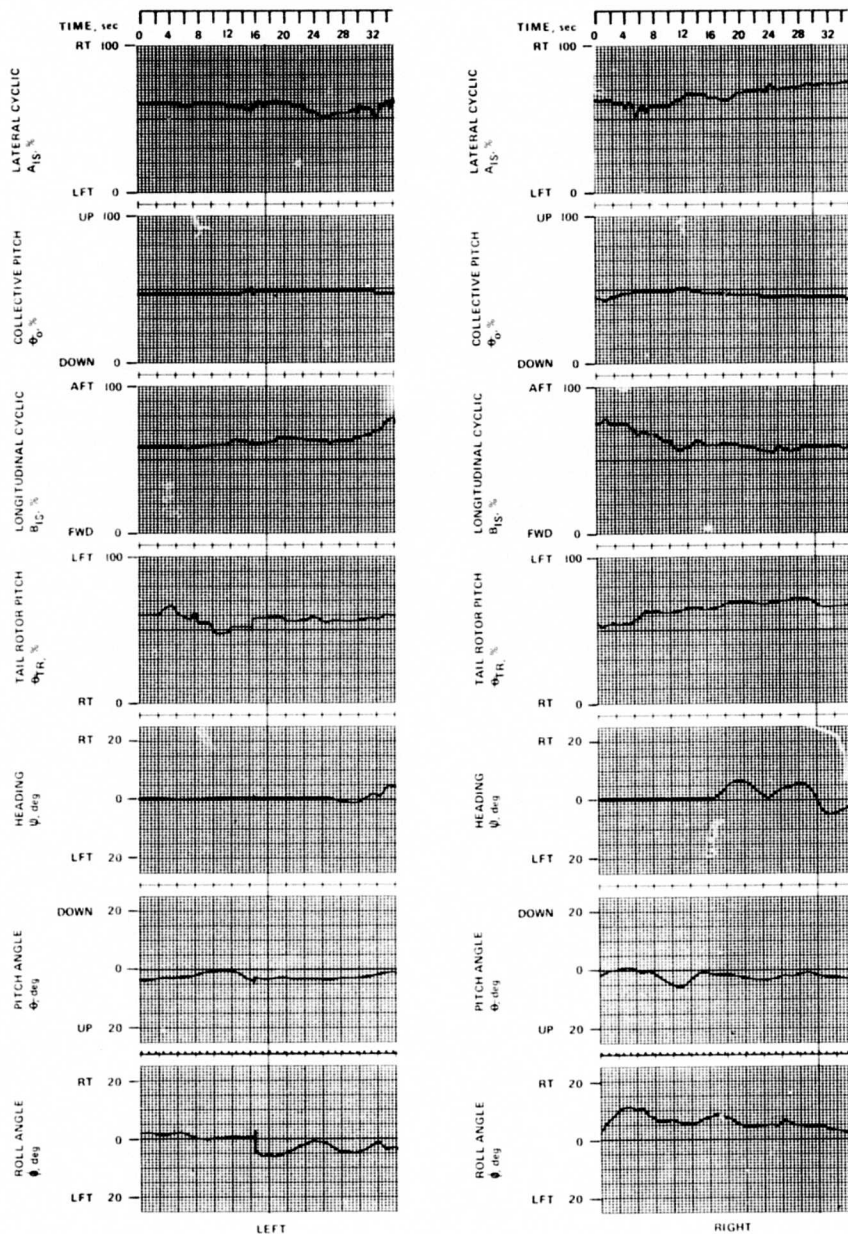


FIGURE 58. AIRCRAFT RESPONSE TO SIDEWARD FLIGHT  
FOR GW = 35,000 LB, CG = 336 IN. AT HOVER.

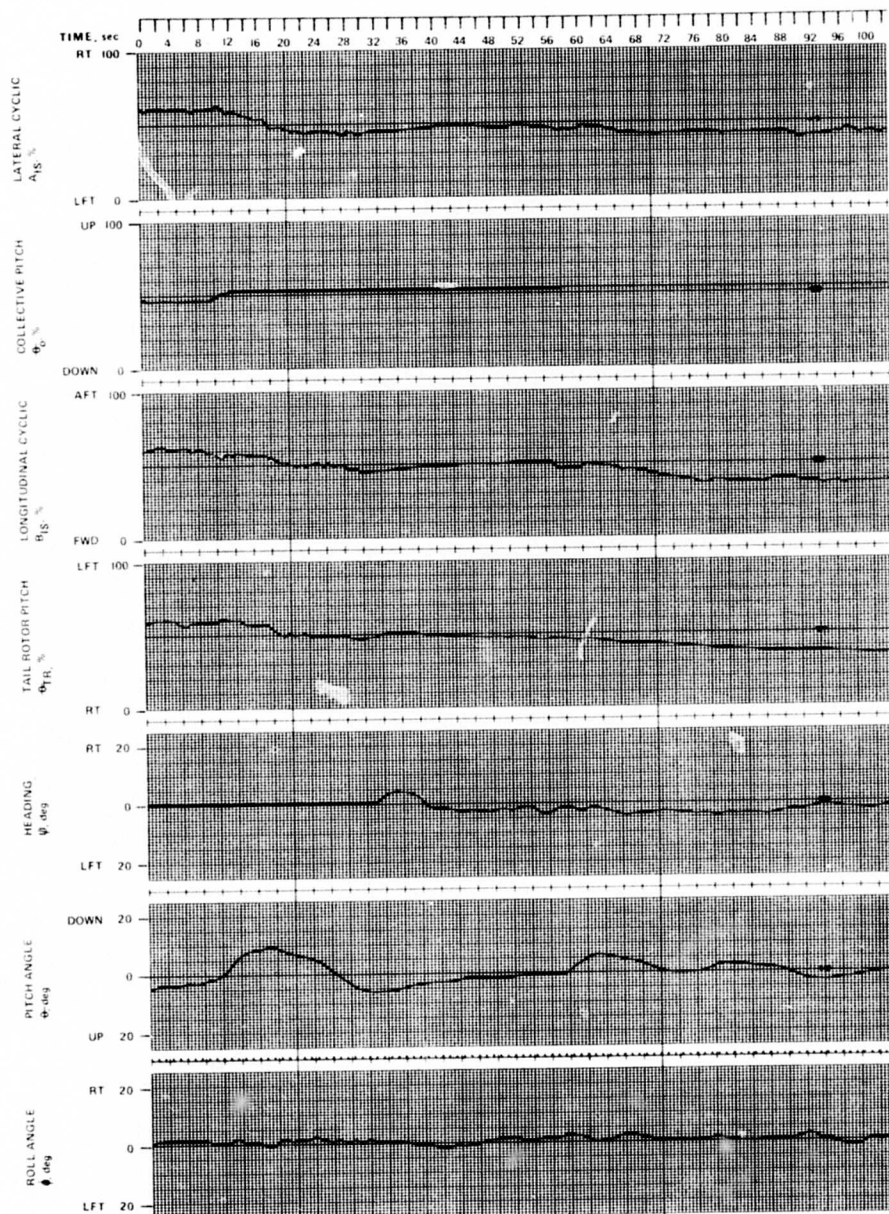


FIGURE 59. AIRCRAFT RESPONSE TO TAKE OFF  
FOR GW = 35,000 LB, CG = 336 IN. AT HOVER.

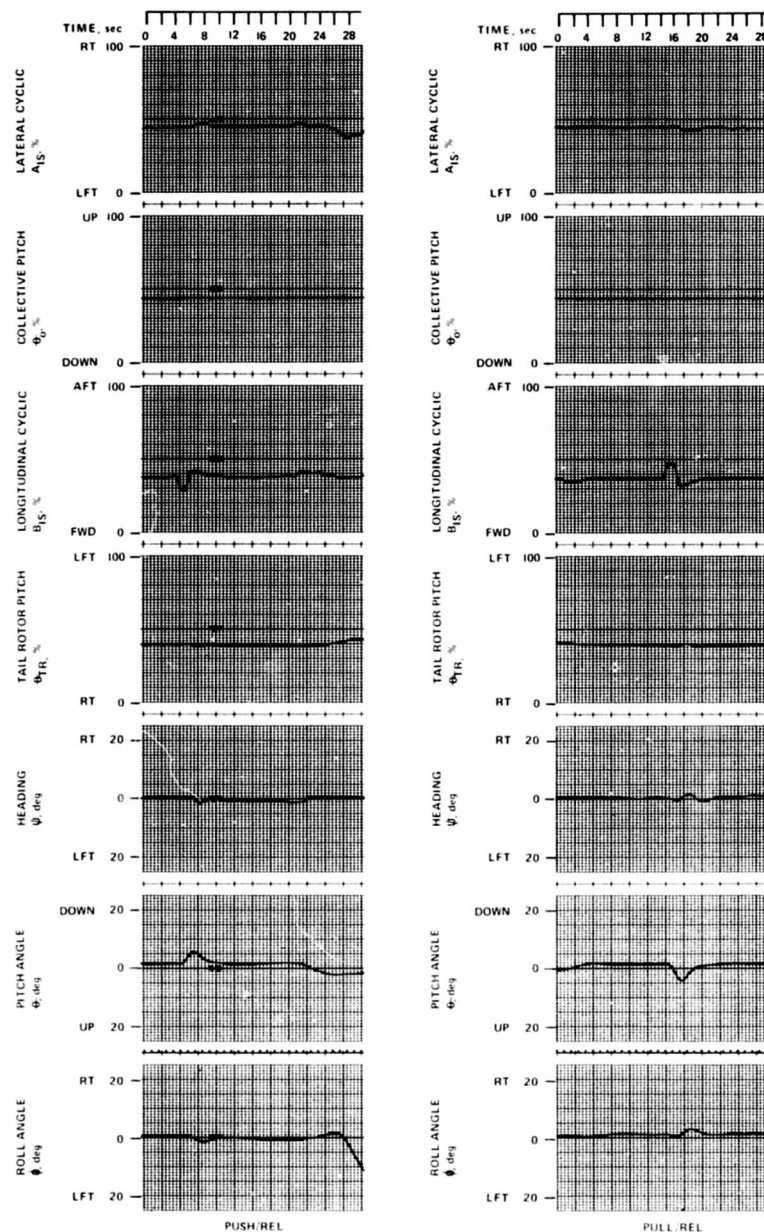


FIGURE 60. AIRCRAFT RESPONSE TO PITCH PULSE INPUTS  
FOR CW = 35,000 LB, CG = 336 IN. AT 80 KNOTS.

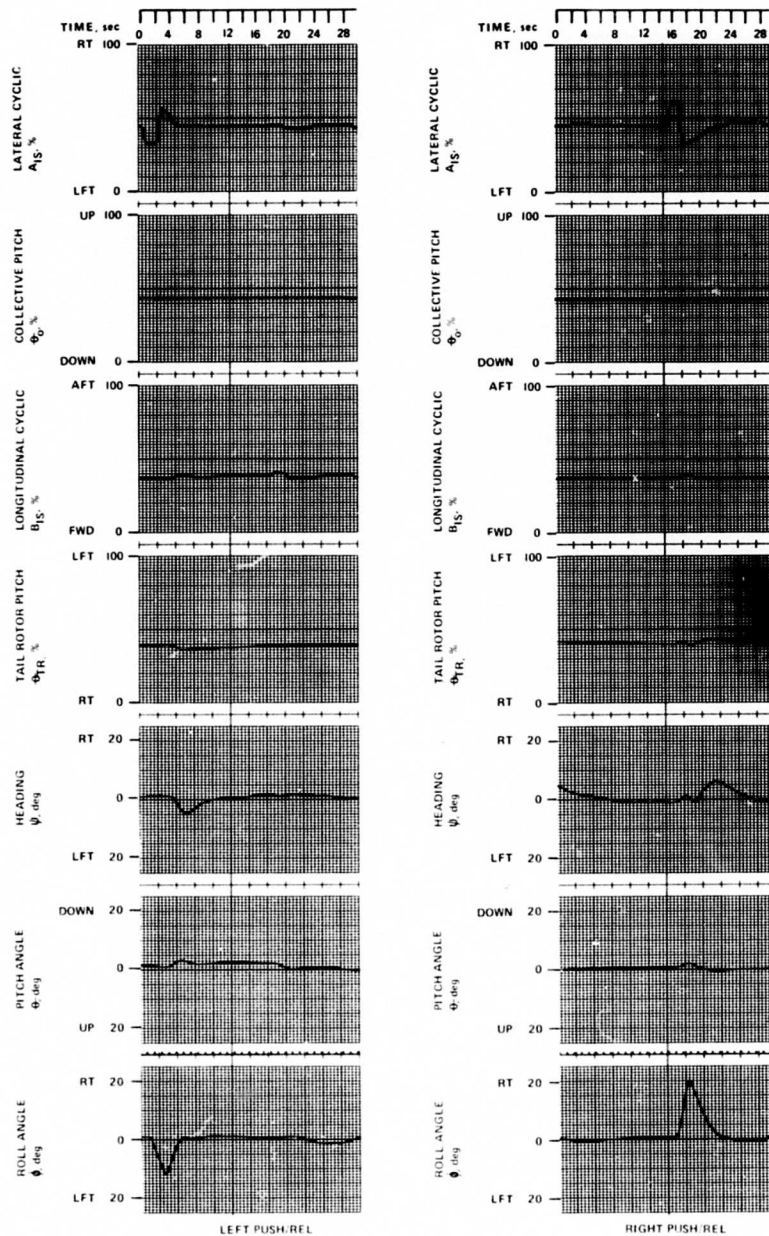


FIGURE 6I. AIRCRAFT RESPONSE TO ROLL PULSE INPUTS  
FOR GW = 35,000 LB, CG = 336 IN. AT 80 KNOTS.

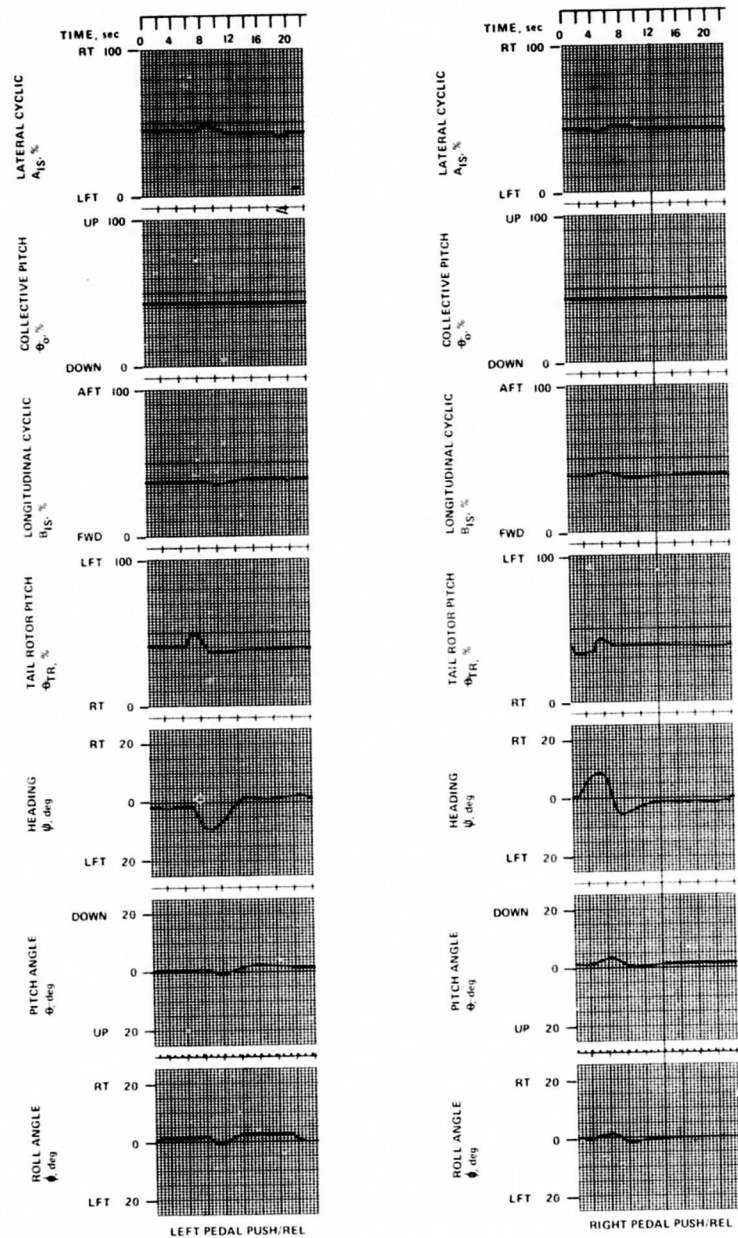
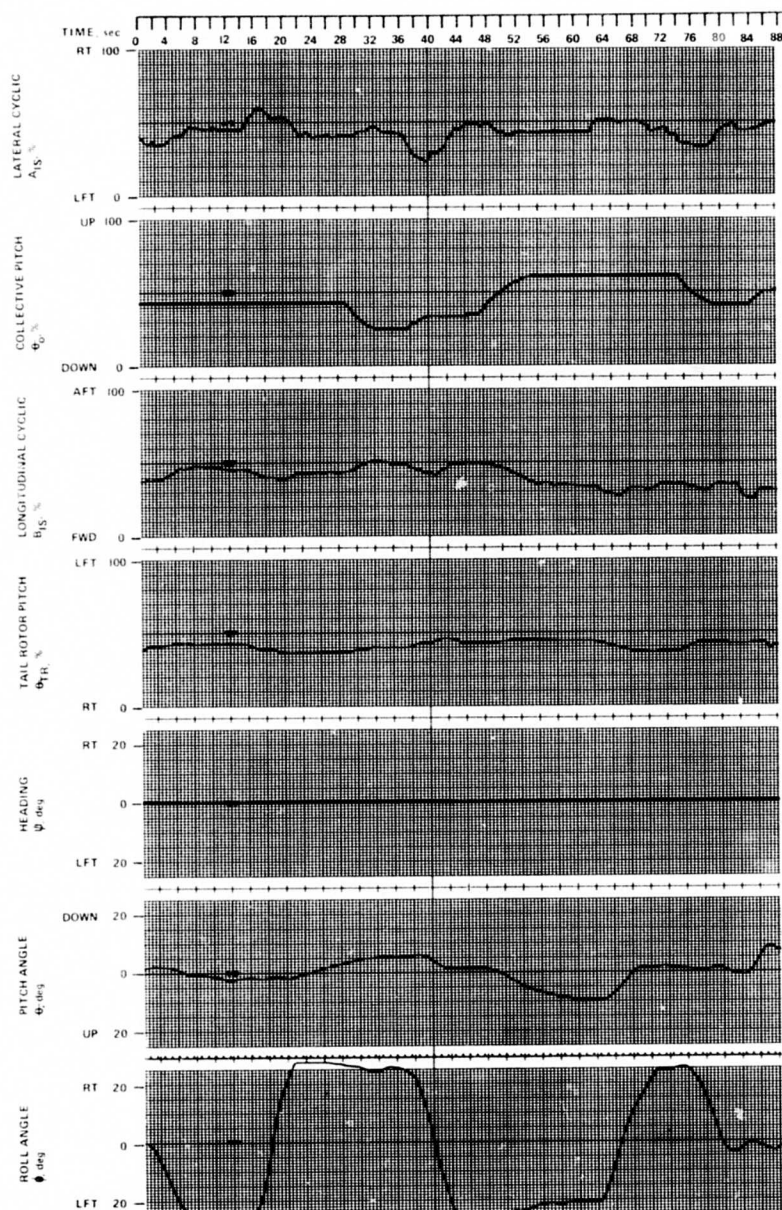
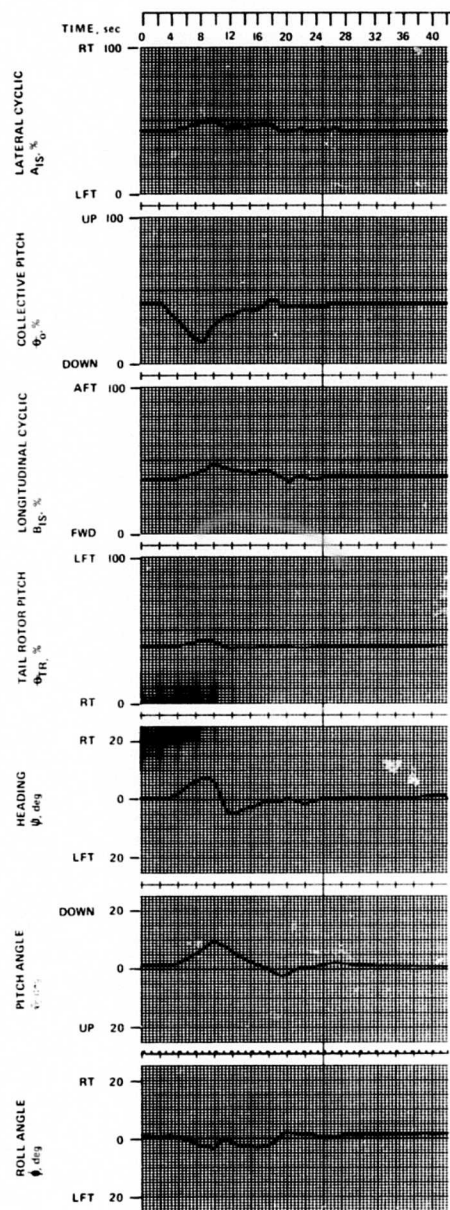


FIGURE 62. AIRCRAFT RESPONSE TO YAW PULSE INPUTS  
FOR GW = 35,000 LB, CG = 336 IN. AT 80 KNOTS.



**FIGURE 63. FORWARD FLIGHT MANEUVERING - CLIMBING AND DESCENDING LEFT AND RIGHT TURNS FOR GW = 35,000 LB, CG = 336 IN. AT 80 KNOTS.**



**FIGURE 64. ENTRY TO AUTOROTATION**  
**FOR GW = 35,000 LB, CG = 336 IN. AT 80 KNOTS.**

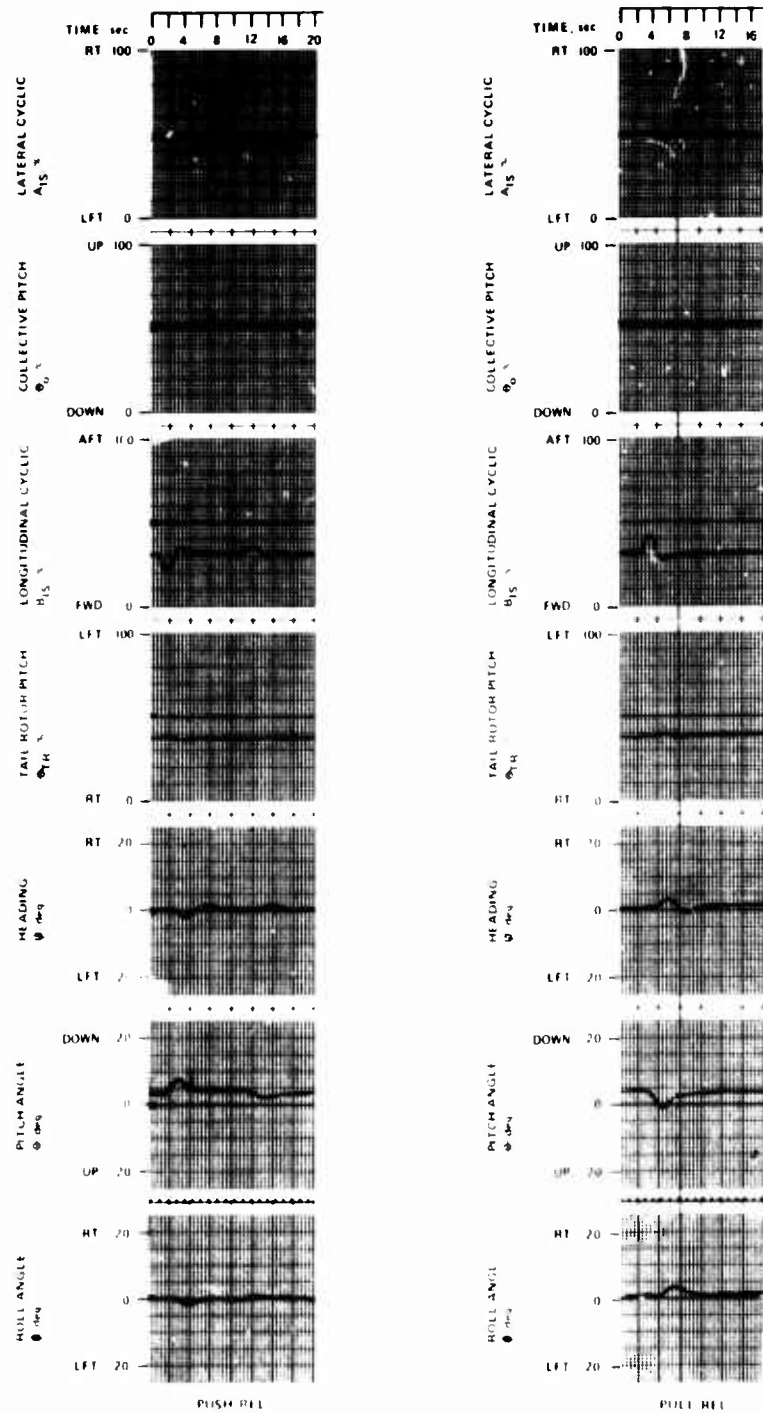


FIGURE 65. AIRCRAFT RESPONSE TO PITCH PULSE INPUTS  
FOR GW = 35,000 LB, CG = 336 IN. AT  $V_{MAX}$ .

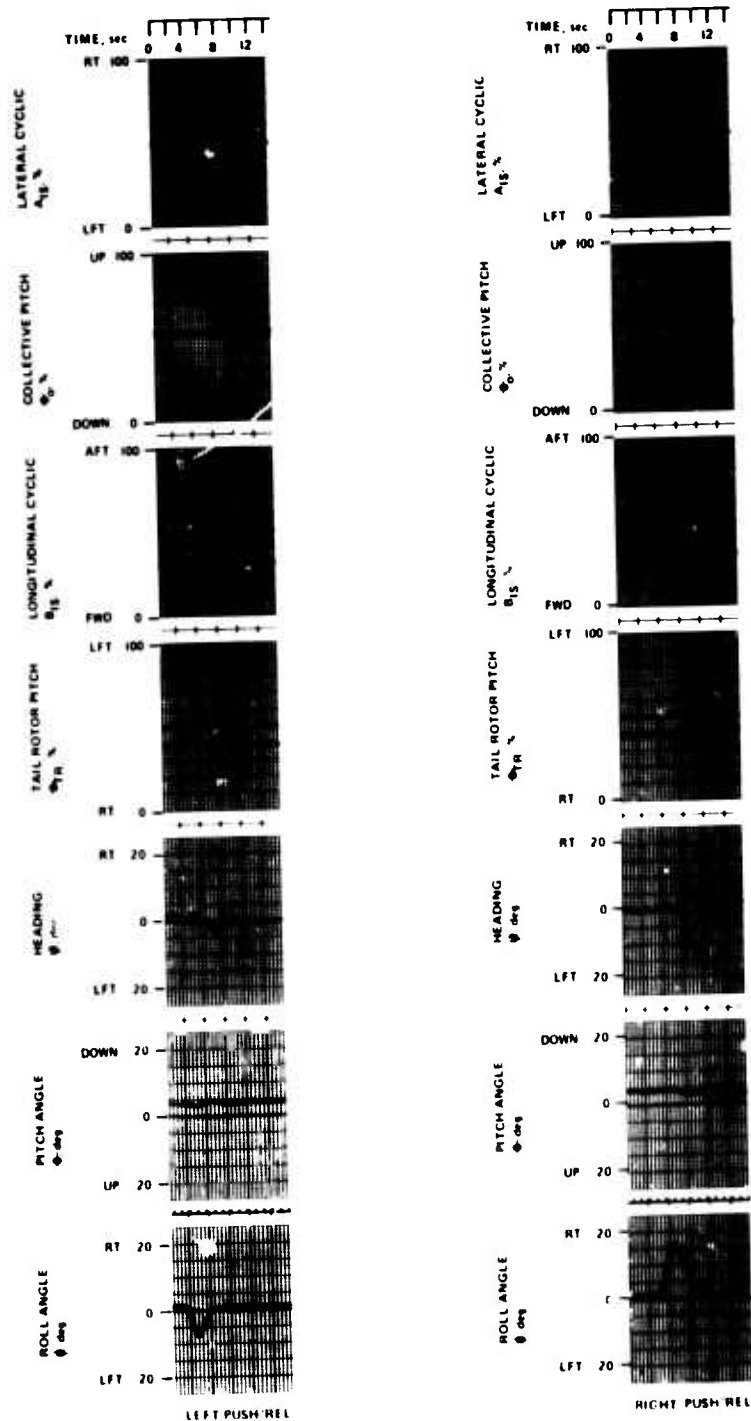


FIGURE 66. AIRCRAFT RESPONSE TO ROLL PULSE INPUTS  
FOR GW = 35,000 LB, CG = 336 IN. AT  $V_{MAX}$ .

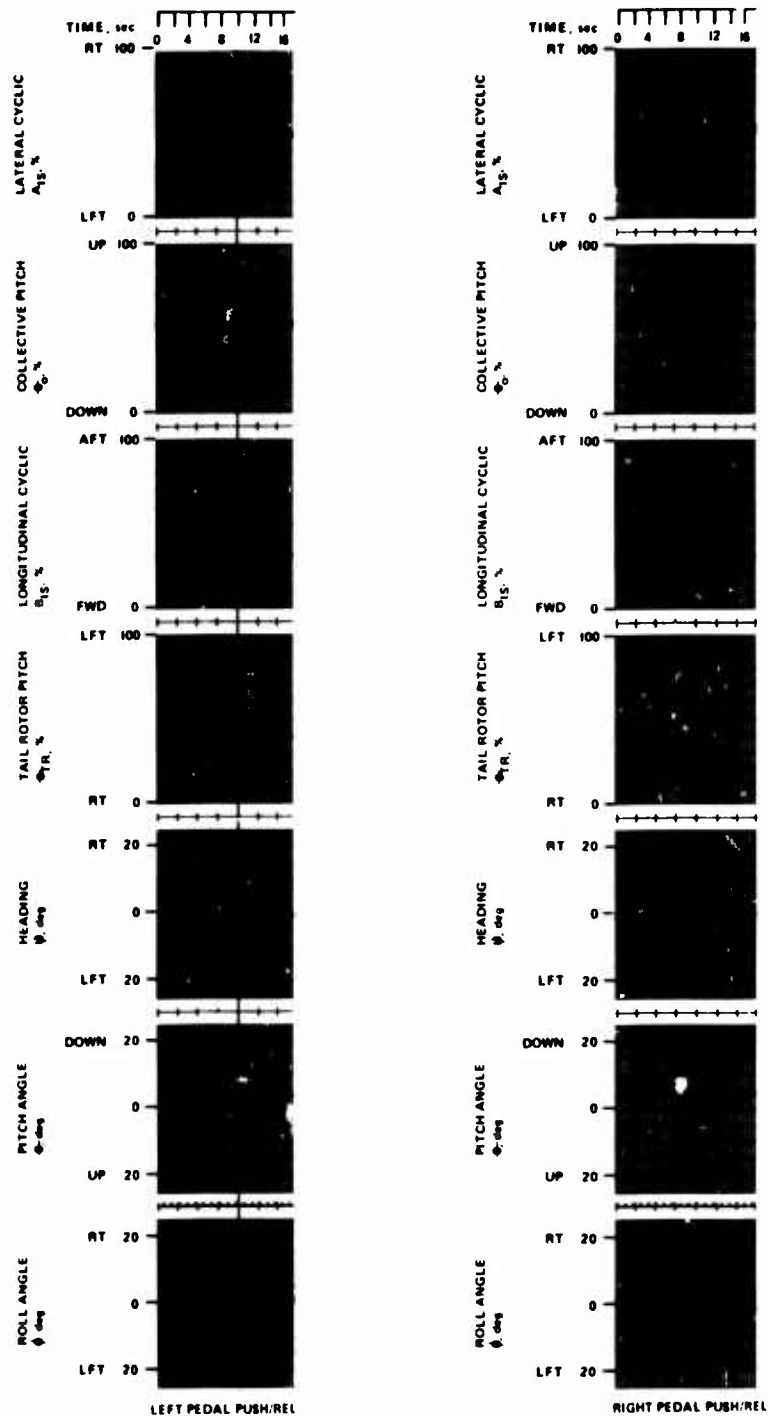


FIGURE 67. AIRCRAFT RESPONSE TO YAW PULSE INPUTS  
FOR GW = 35,000 LB, CG = 336 IN. AT  $V_{MAX}$ .

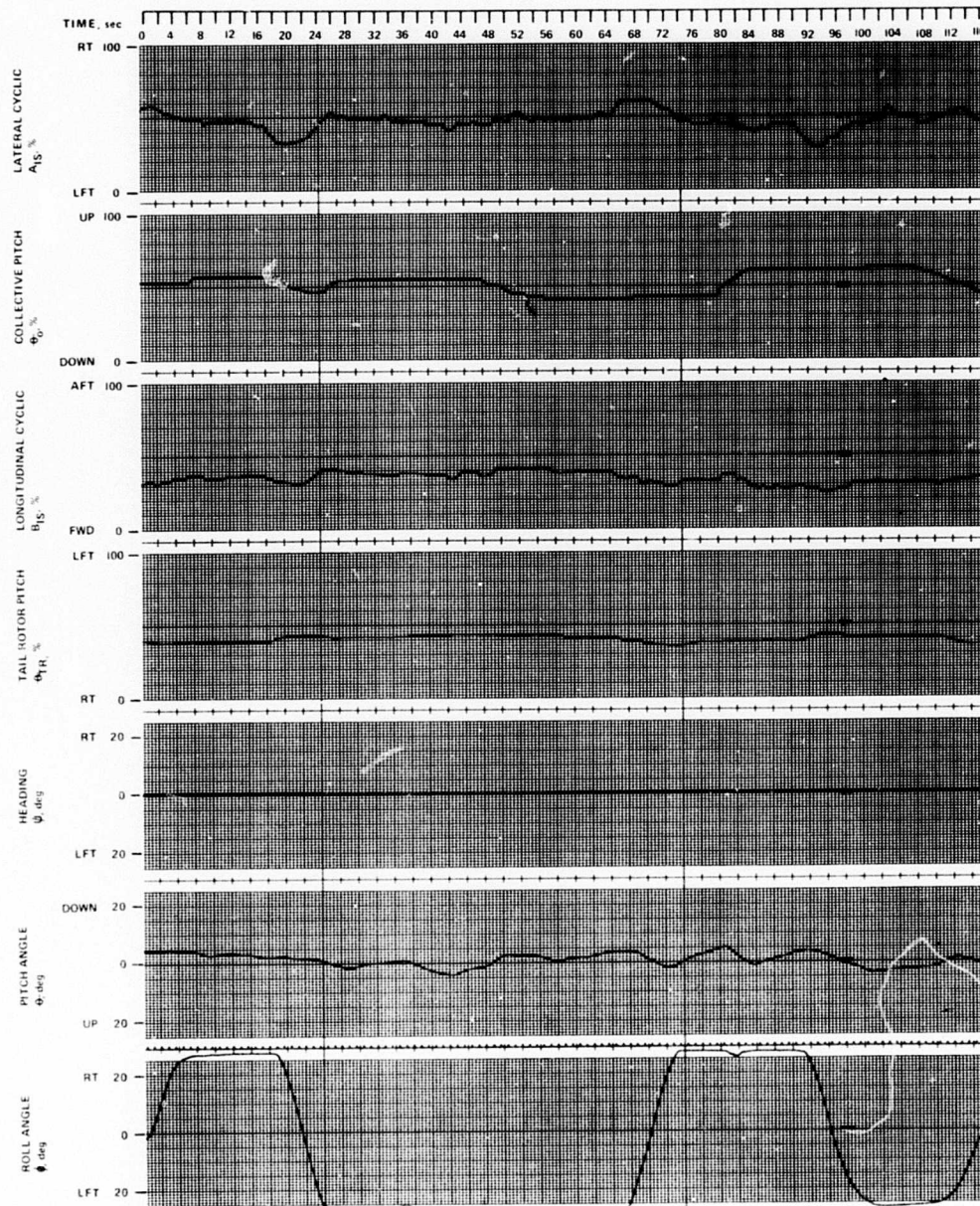
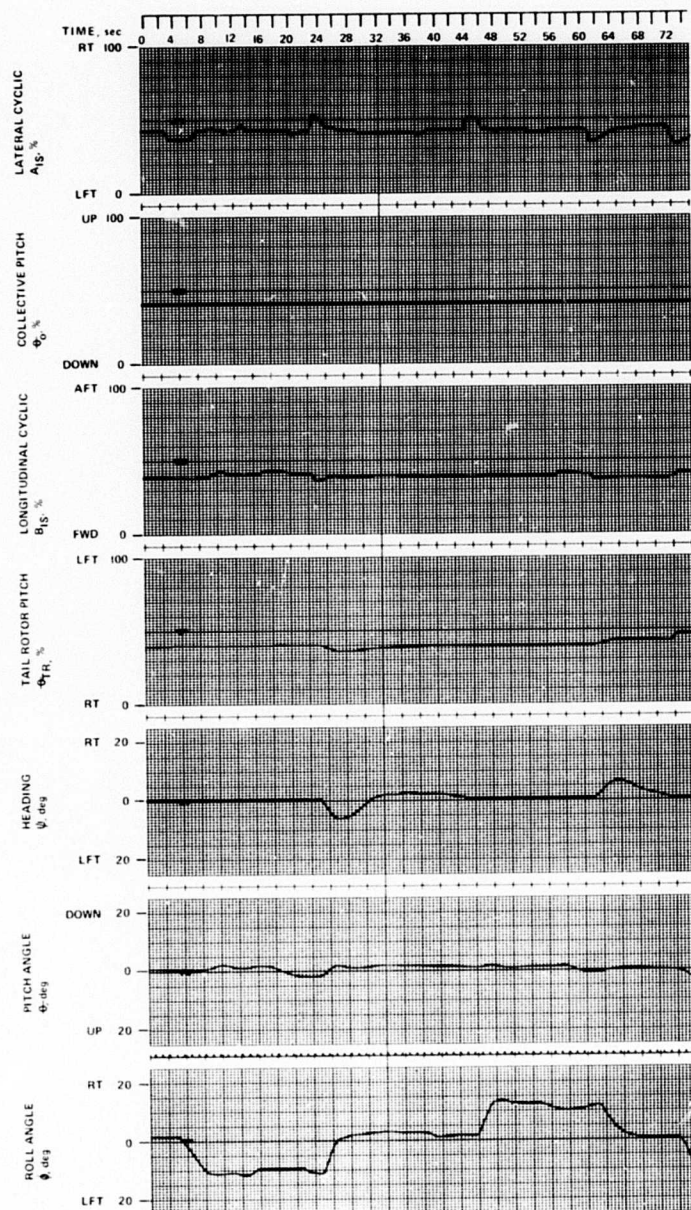
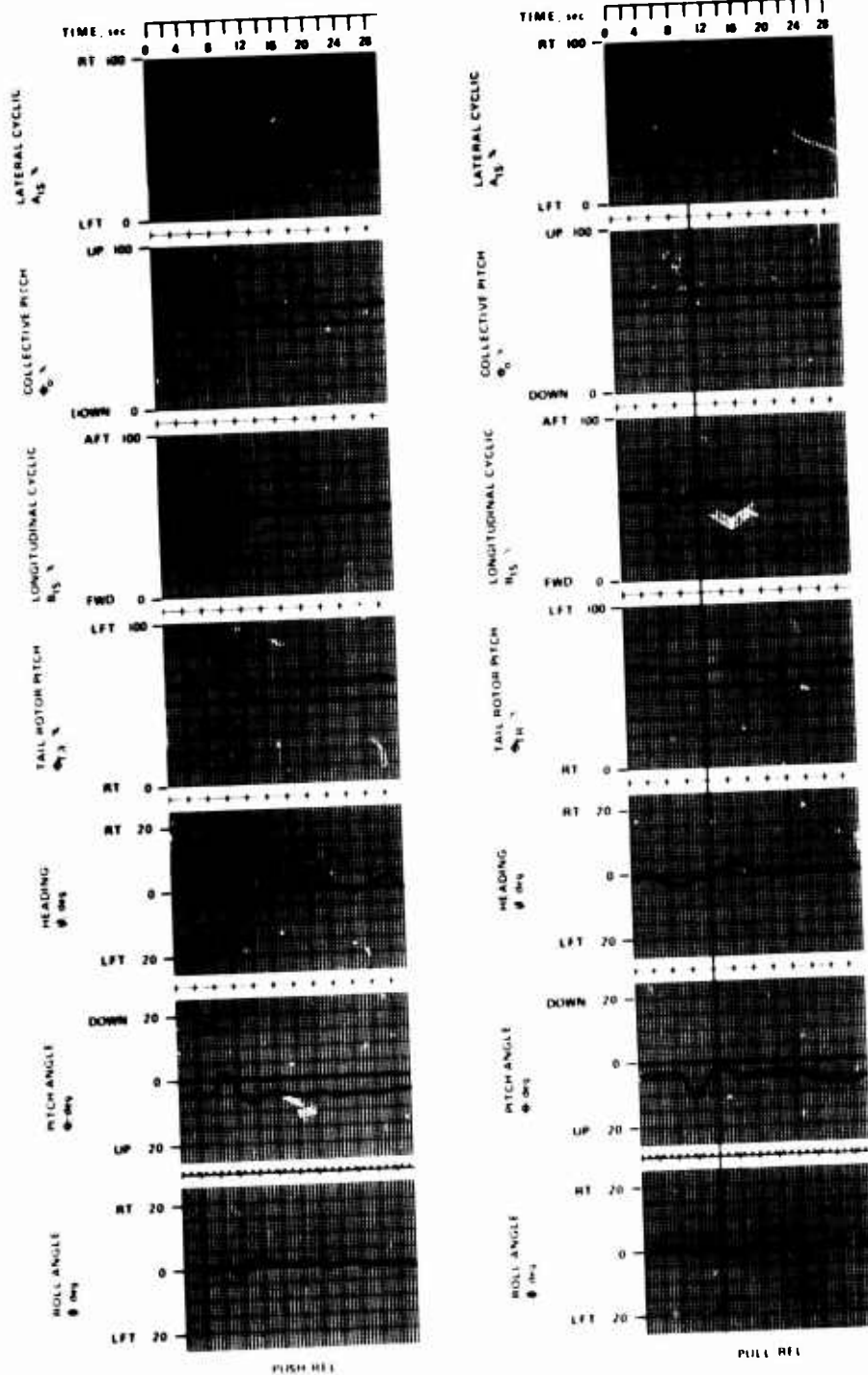


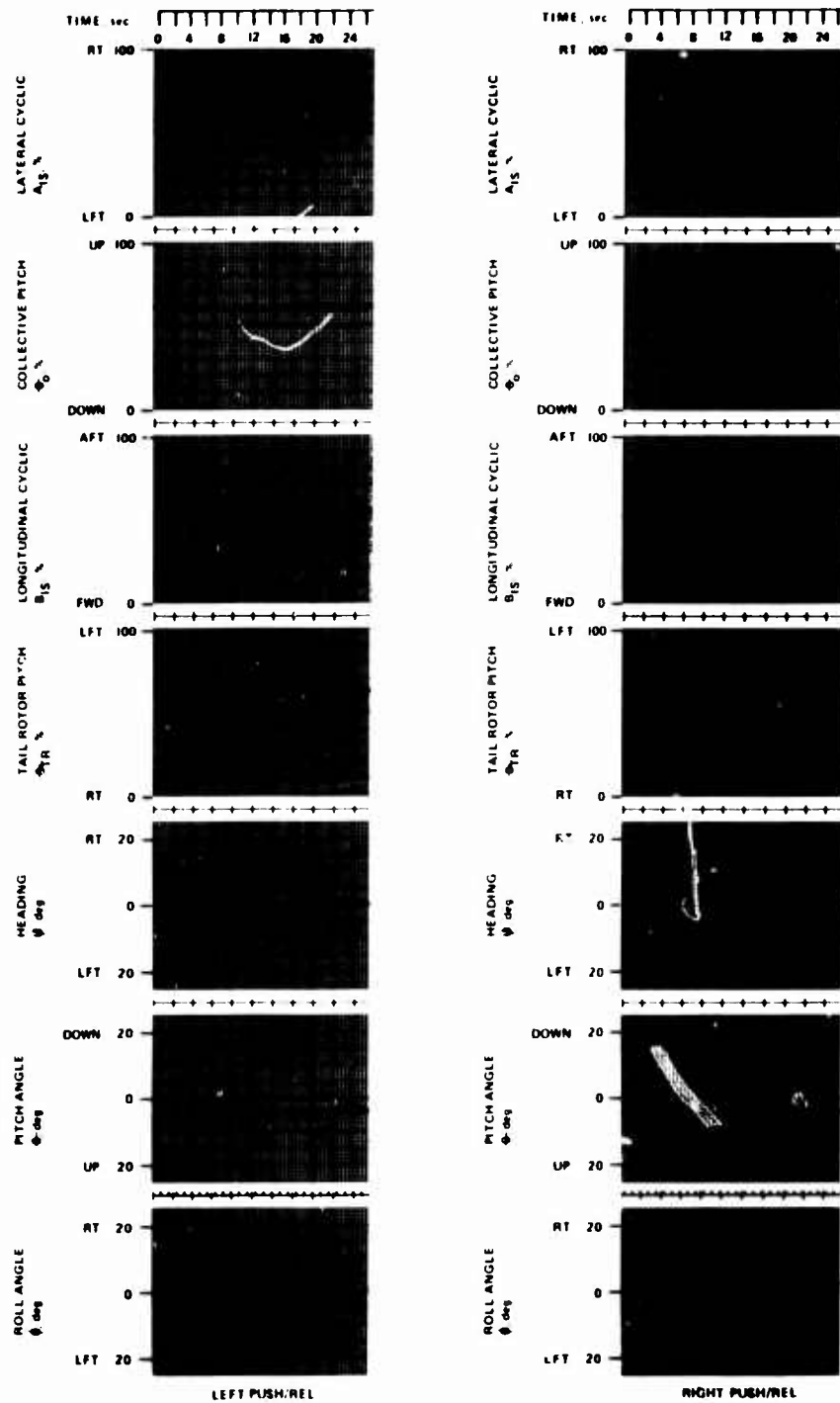
FIGURE 68. FORWARD FLIGHT MANEUVERING - CLIMBING AND DESCENDING  
LEFT AND RIGHT TURNS FOR GW = 35,000 LB, CG = 336 IN. AT  $V_{MAX}$



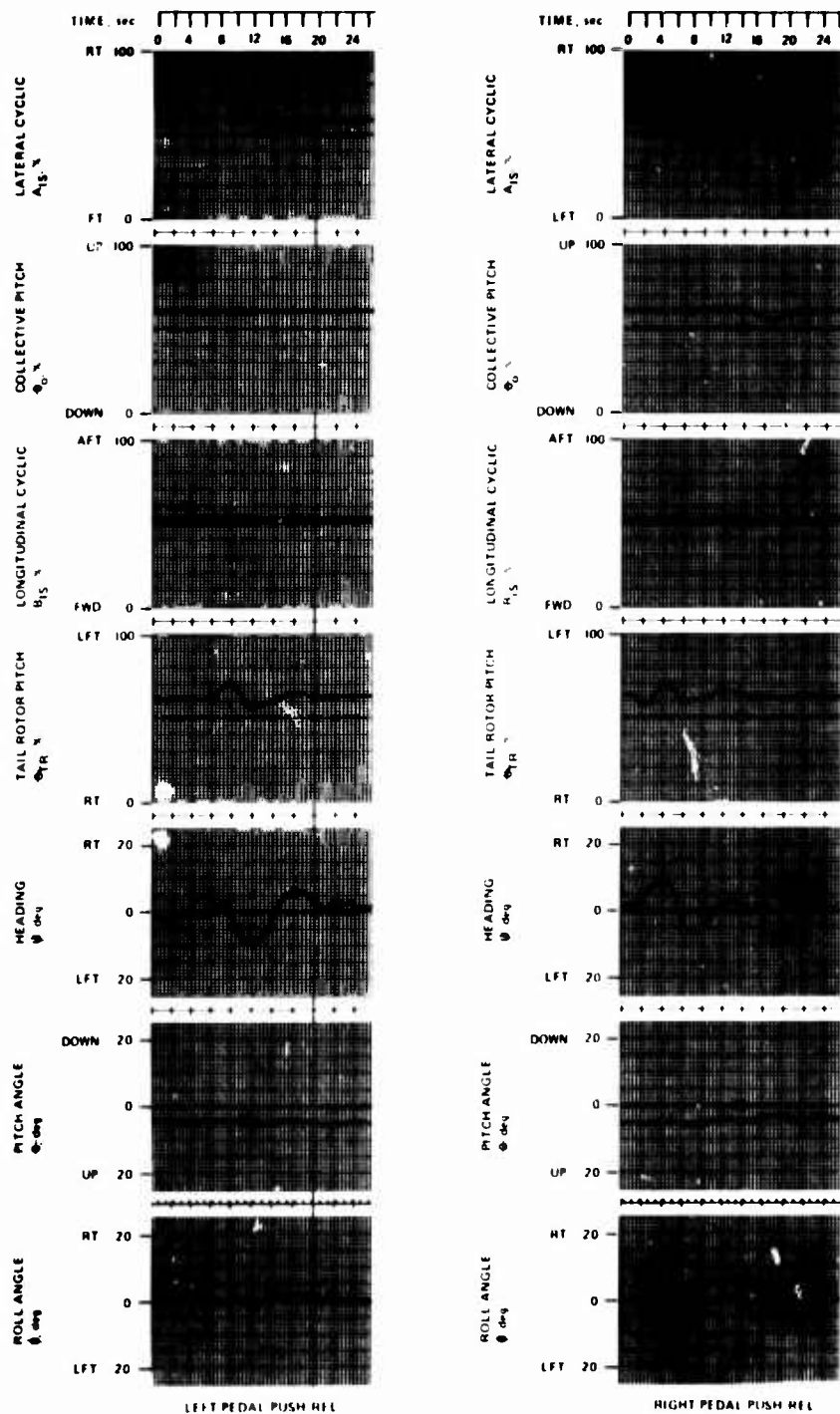
**FIGURE 69. TURNS AGAINST TRIM WITH RELEASE OF ALL CONTROLS  
FOR GW = 35,000 LB, CG = 336 IN. AT  $V_{MAX}$**



**FIGURE 70. AIRCRAFT RESPONSE TO PITCH PULSE INPUTS  
FOR GW = 47,000 LB, CG = 346 IN. AT HOVER.**



**FIGURE 71. AIRCRAFT RESPONSE TO ROLL PULSE INPUTS  
FOR GW = 47,000 LB, CG = 346 IN. AT HOVER.**



**FIGURE 72. AIRCRAFT RESPONSE TO YAW PULSE INPUTS  
FOR GW = 47,000 LB, CG = 346 IN. AT HOVER.**

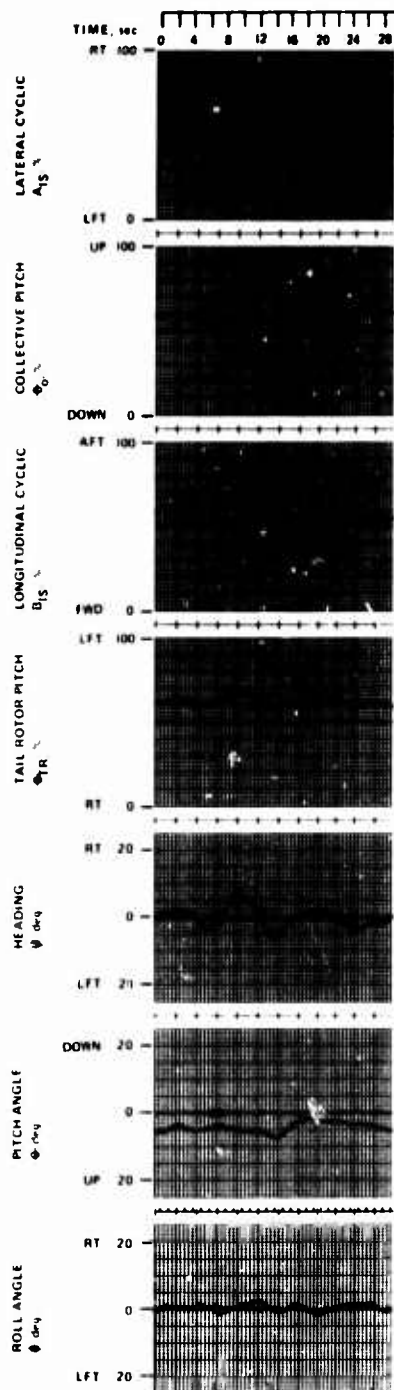
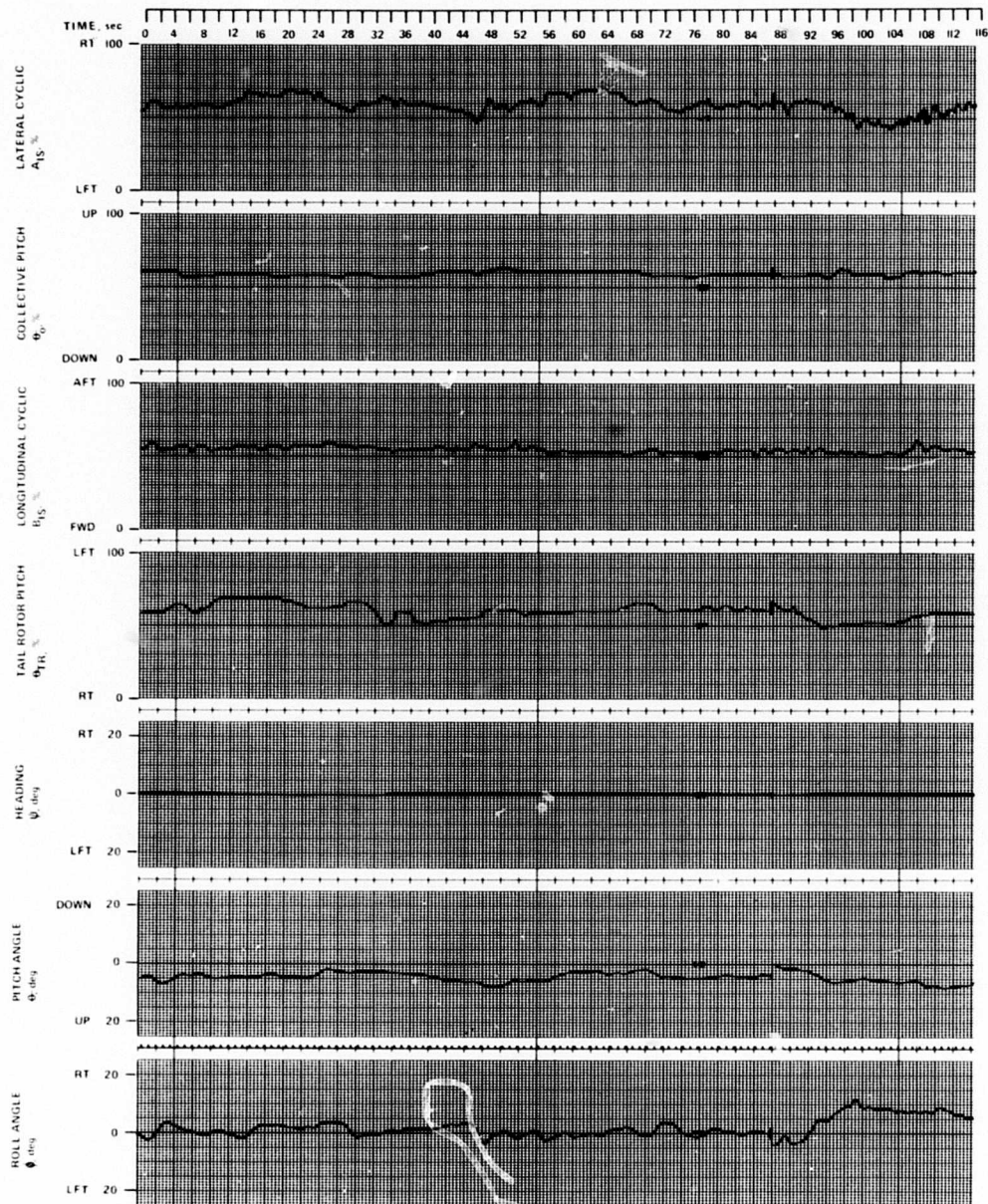


FIGURE 73. AIRCRAFT RESPONSE TO COLLECTIVE STEP INPUTS  
FOR GW = 47,000 LB, CG = 346 IN. AT HOVER.



**FIGURE 74. HOVER MANEUVERING FLIGHT - SLOW FLIGHT AND S TURNS  
FOR GW = 47,000 LB, CG = 346 IN. AT HOVER.**

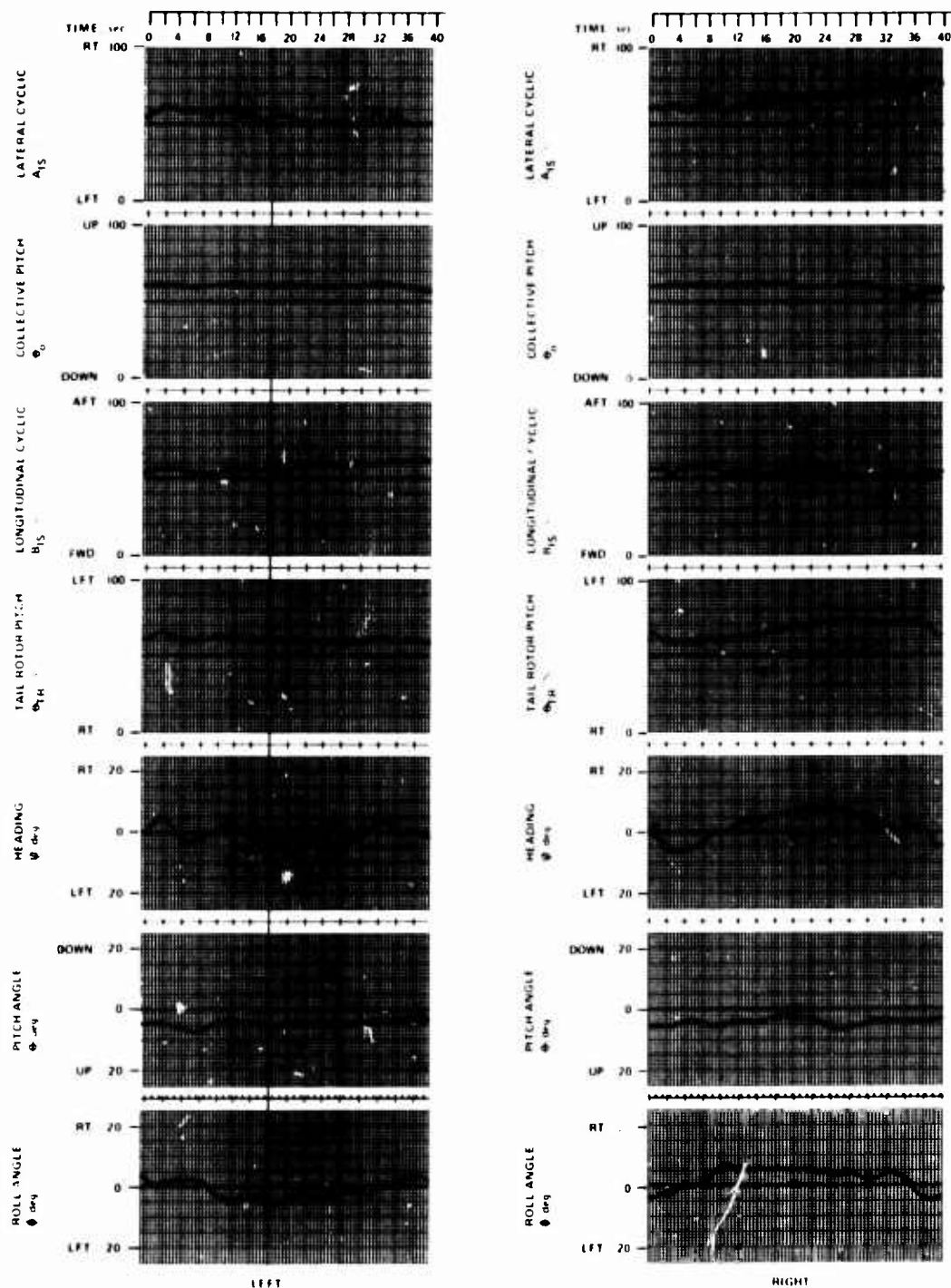
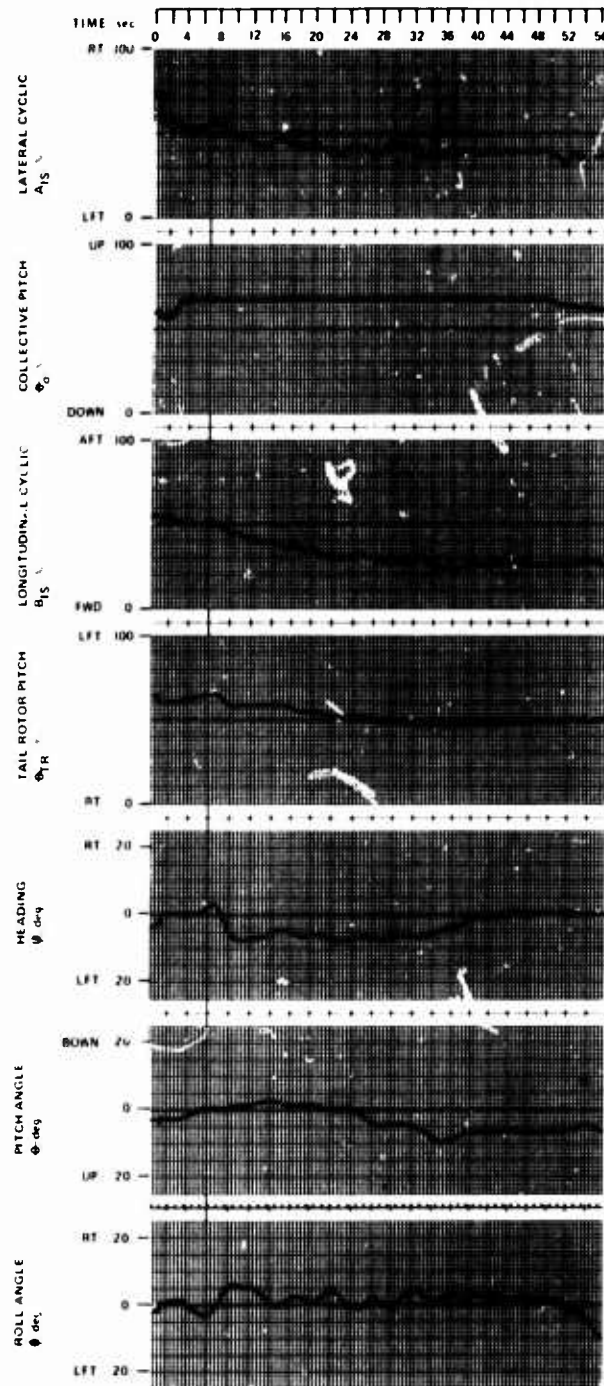
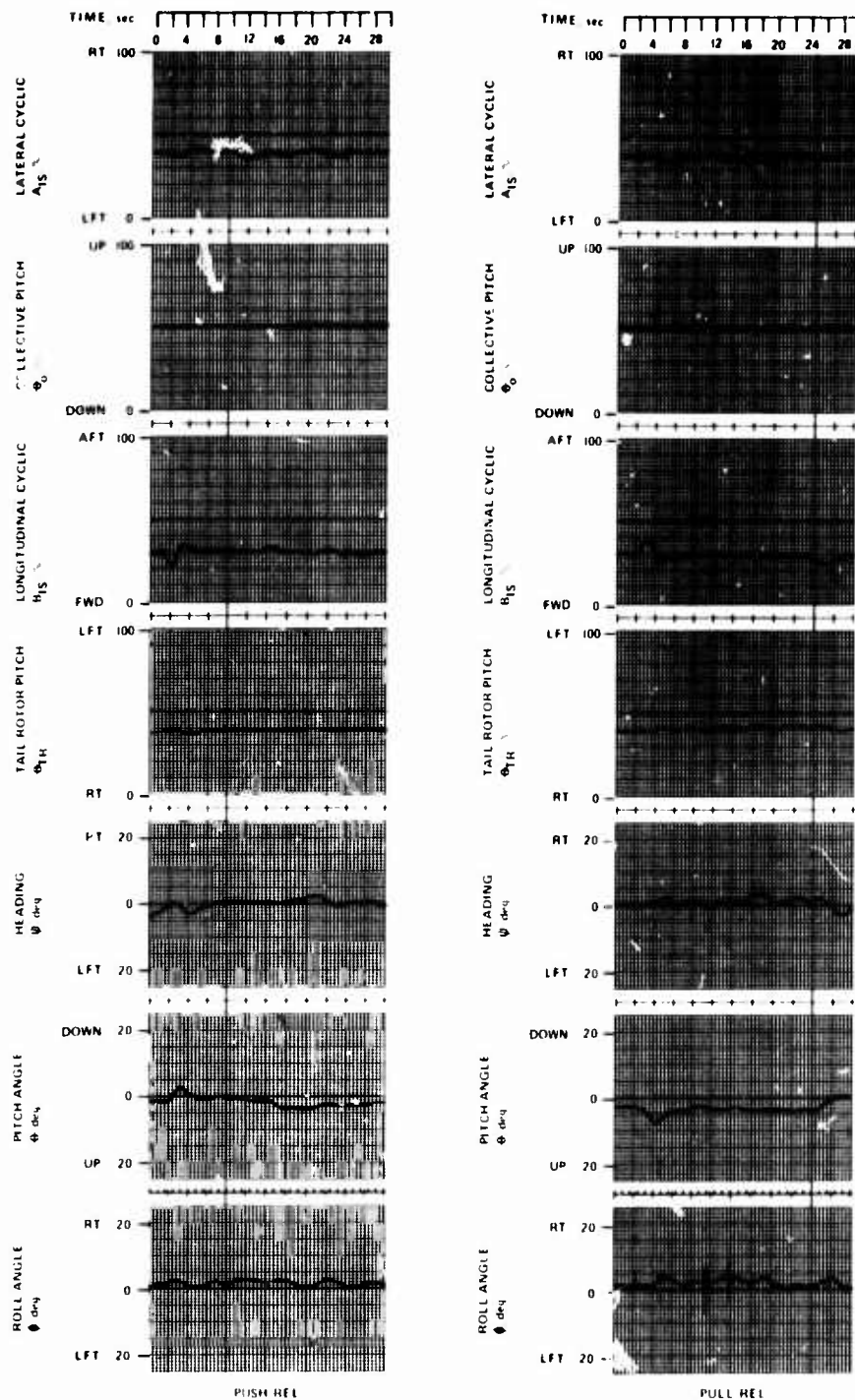


FIGURE 75. AIRCRAFT RESPONSE TO SIDEWARD FLIGHT  
FOR  $GW = 47,000 \text{ LB}$ ,  $CG = 346 \text{ IN. AT HOVER}$ .



**FIGURE 76. AIRCRAFT RESPONSE TO TAKEOFF  
FOR GW = 47,000 LB, CG = 340 IN. AT HOVER.**



**FIGURE 77. AIRCRAFT RESPONSE TO PITCH PULSE INPUTS  
FOR GW = 47,000 LB, CG = 346 IN. AT 80 KNOTS.**

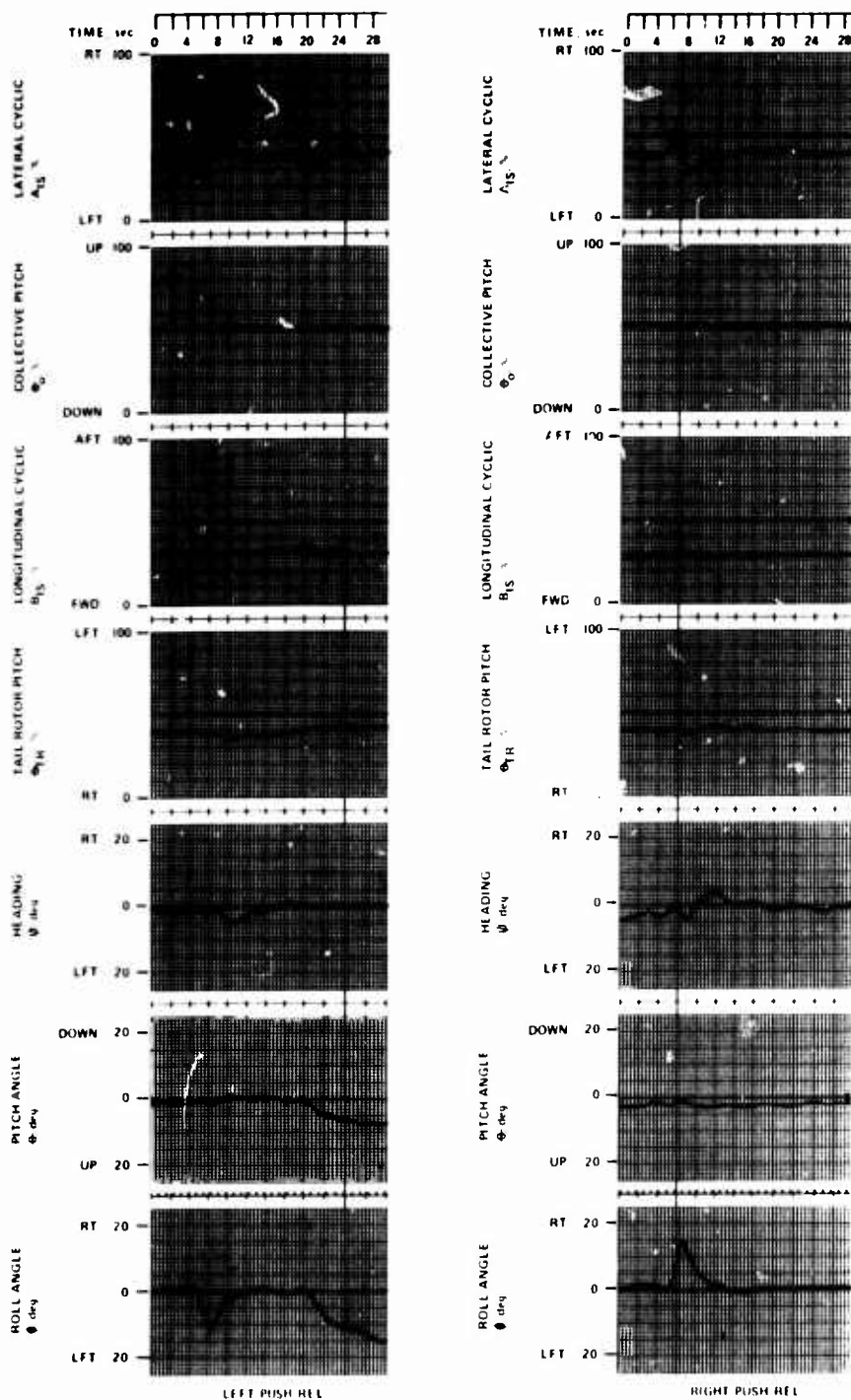


FIGURE 78. AIRCRAFT RESPONSE TO ROLL PULSE INPUTS  
FOR GW = 47,000 LB, CG = 346 IN. AT 80 KNOTS.

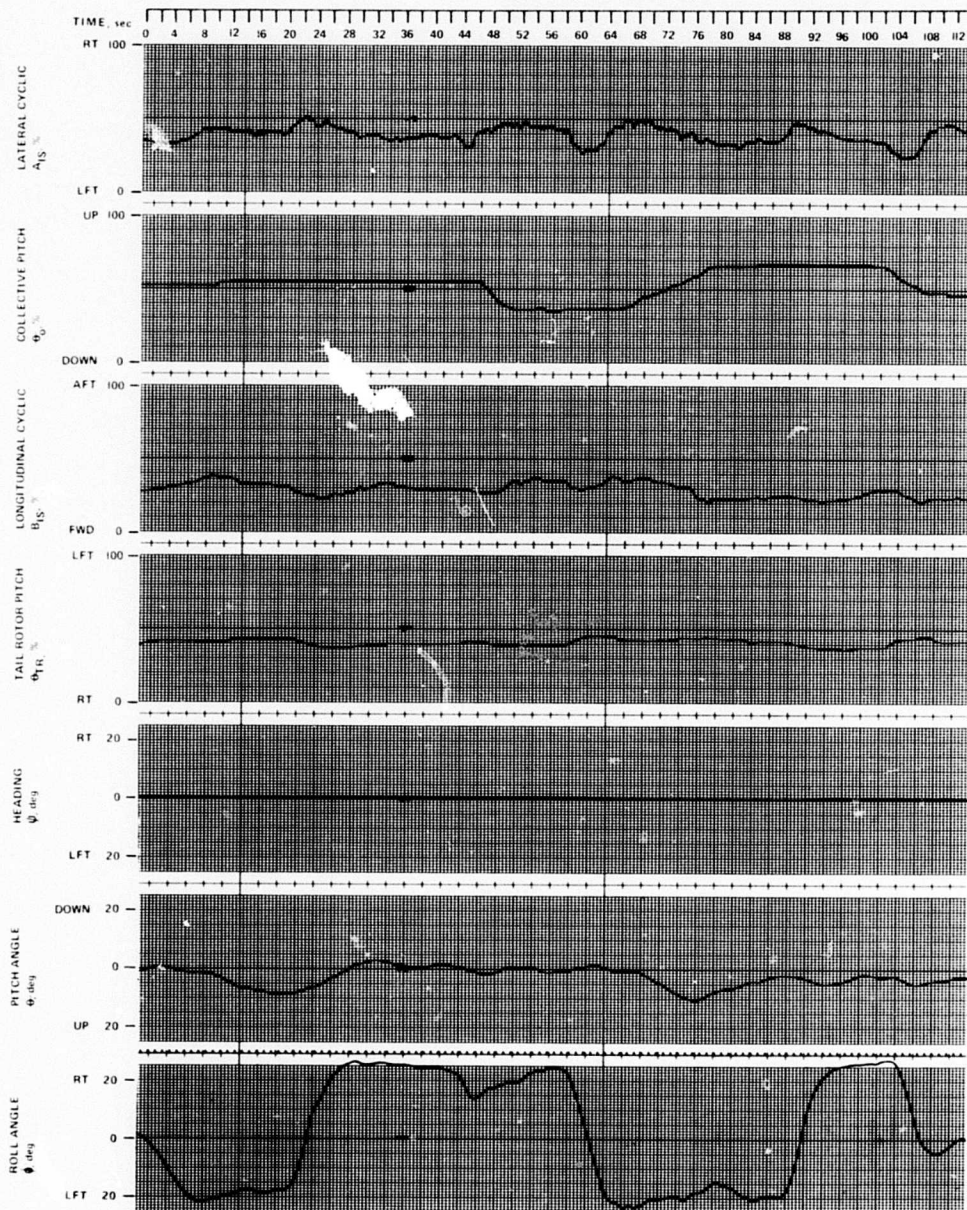
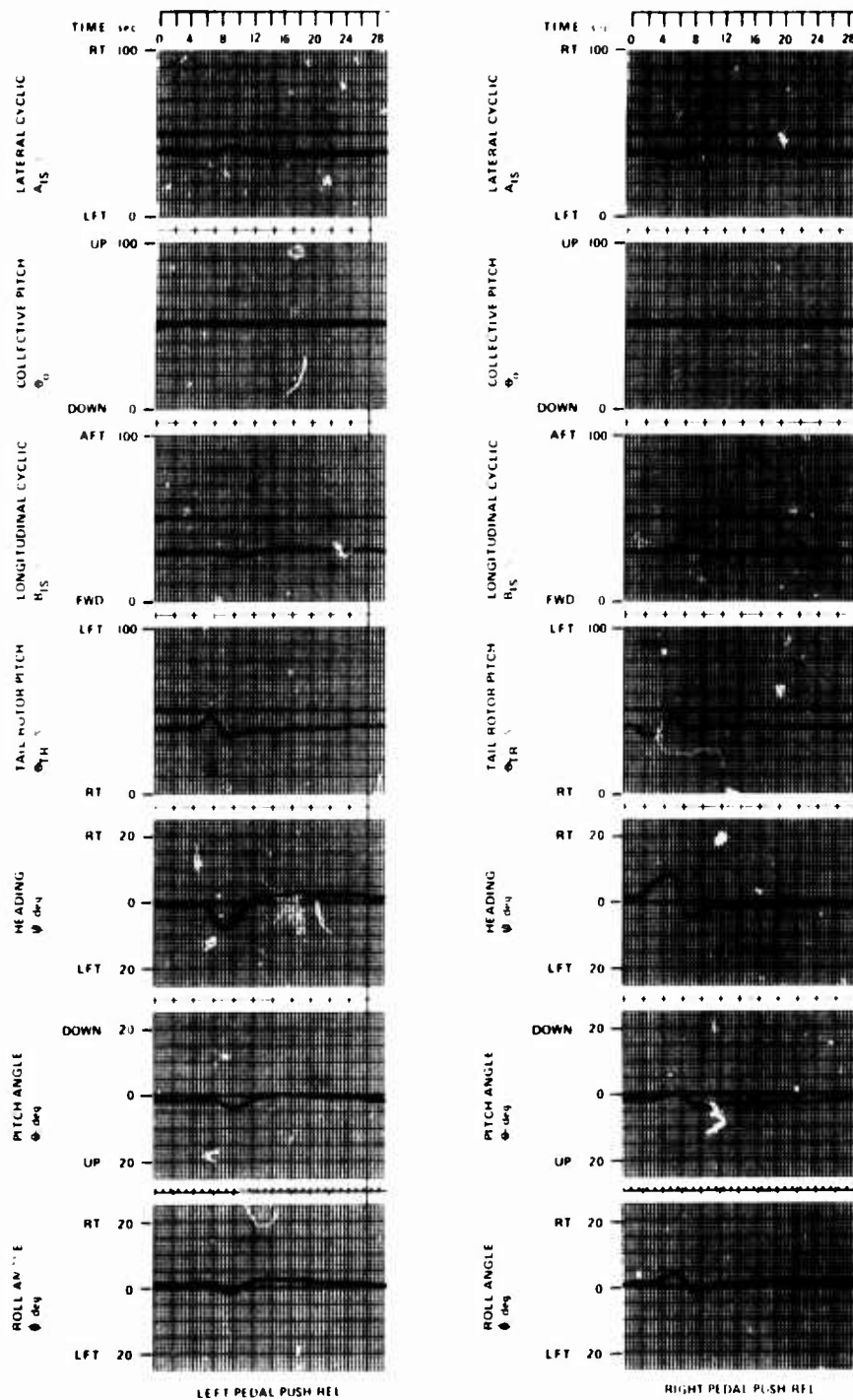
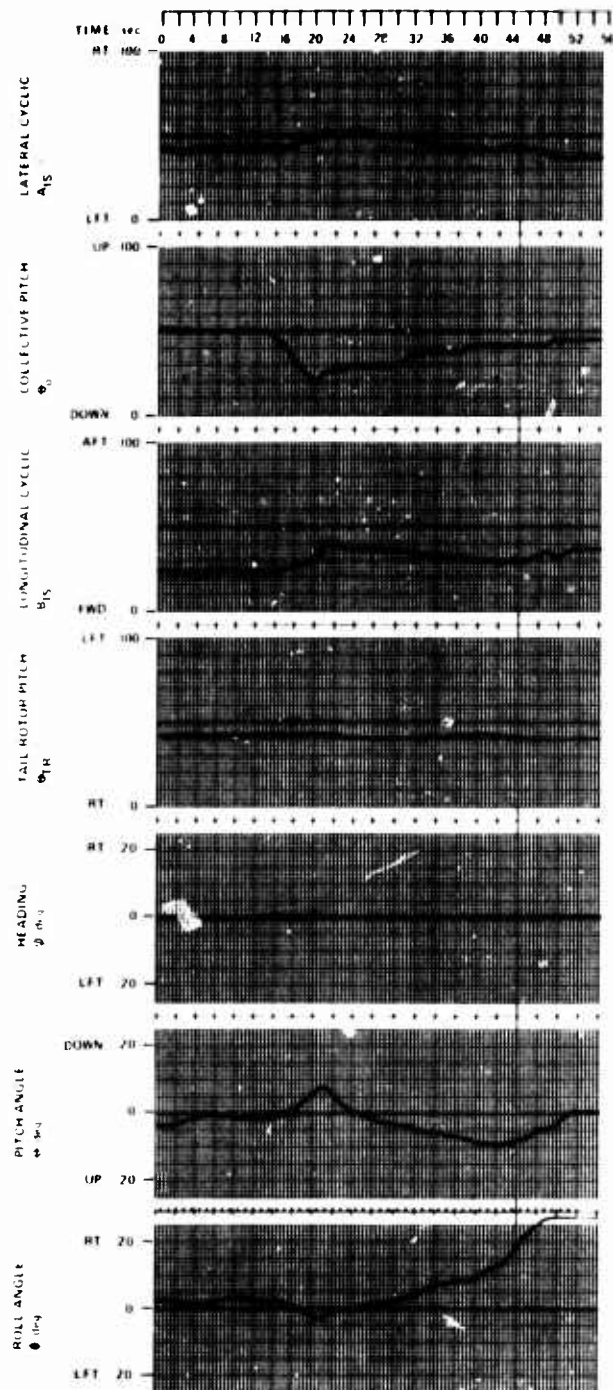


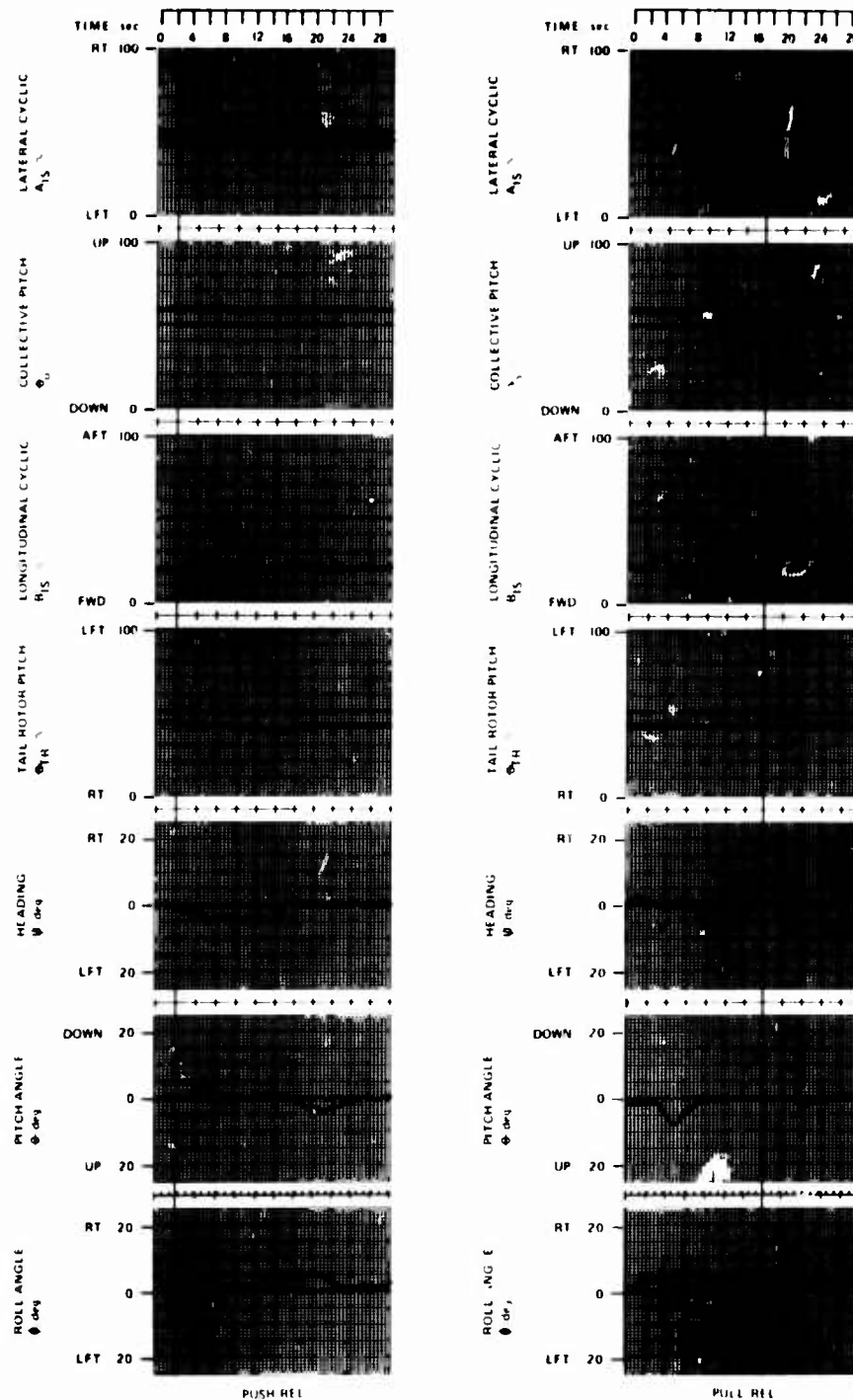
FIGURE 79. AIRCRAFT RESPONSE TO YAW PULSE INPUTS  
FOR GW = 47,000 LB, CG = 346 IN. AT 80 KNOTS.



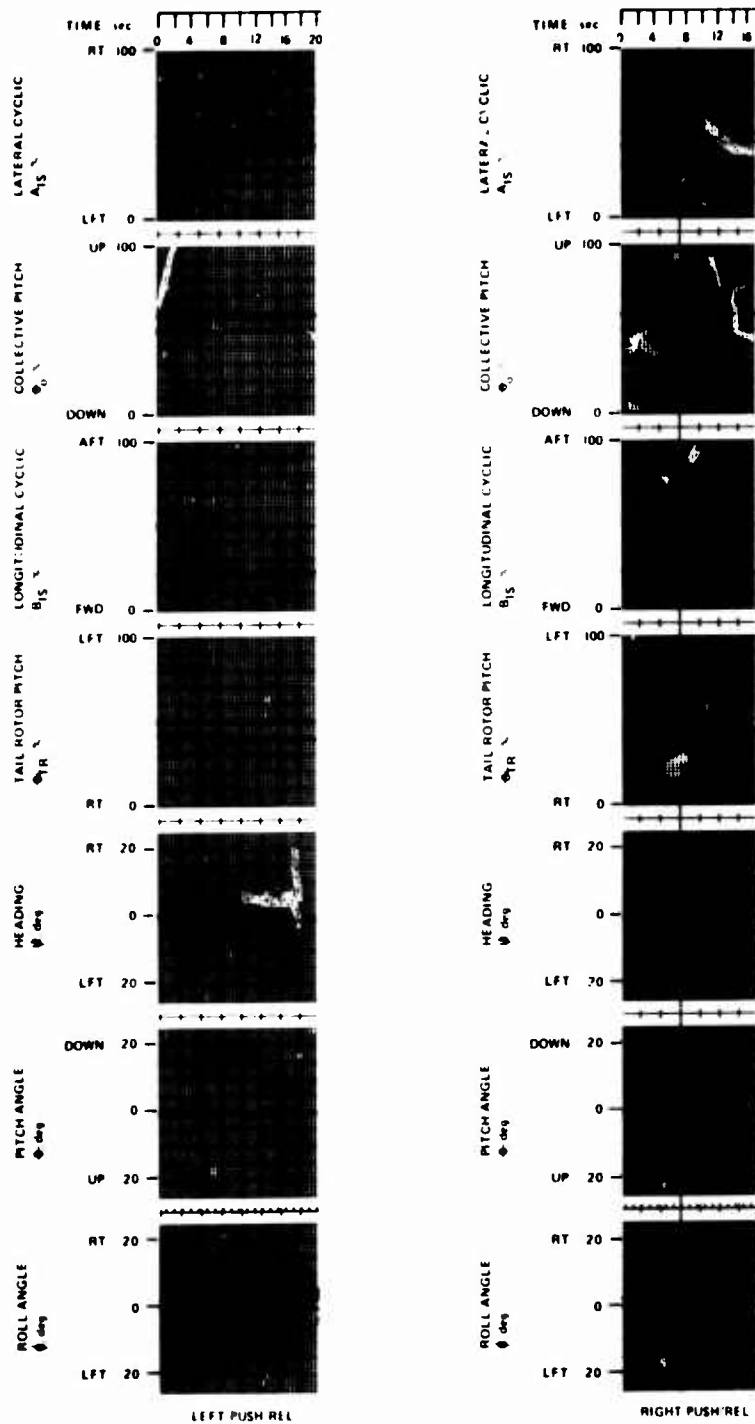
**FIGURE 80. FORWARD FLIGHT MANEUVERING - CLIMBING AND DESCENDING LEFT AND RIGHT TURNS FOR GW = 47,000 LB, CG = 346 IN. AT 80 KNOTS.**



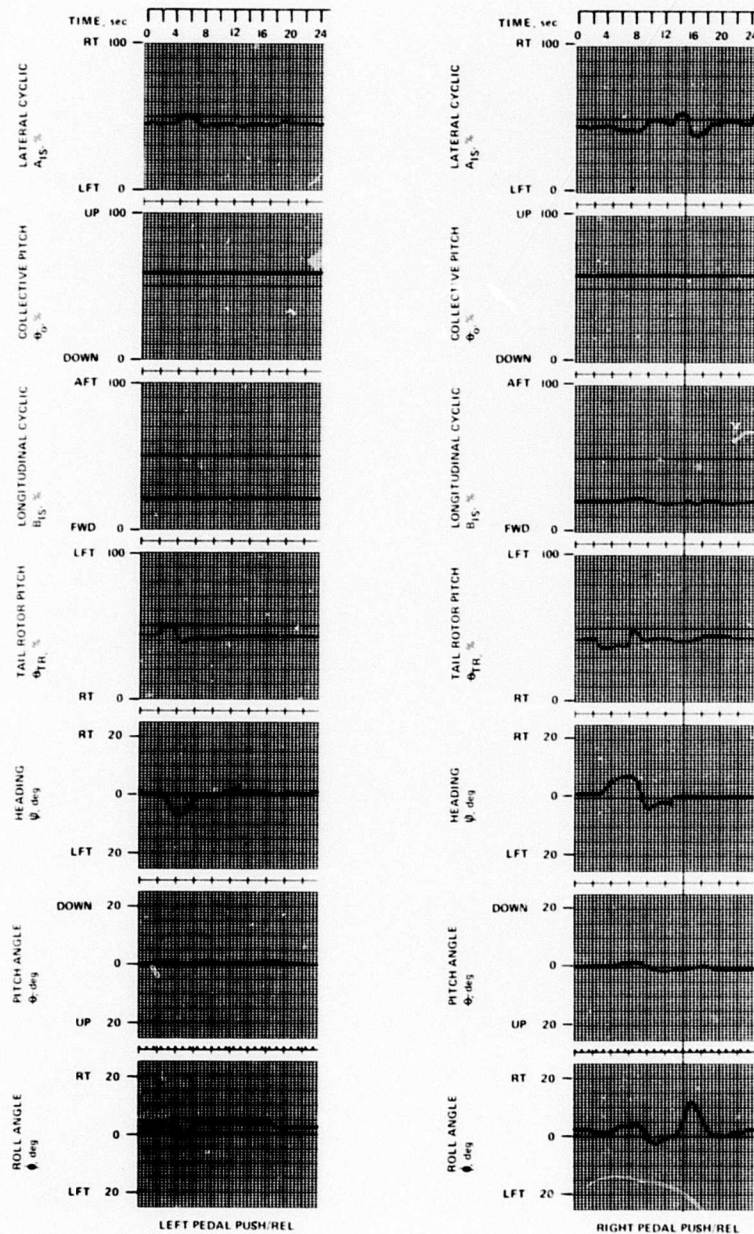
**FIGURE 8I. ENTRY TO AUTOROTATION  
FOR GW = 47,000 LB, CG = 346 IN. AT 80 KNOTS.**



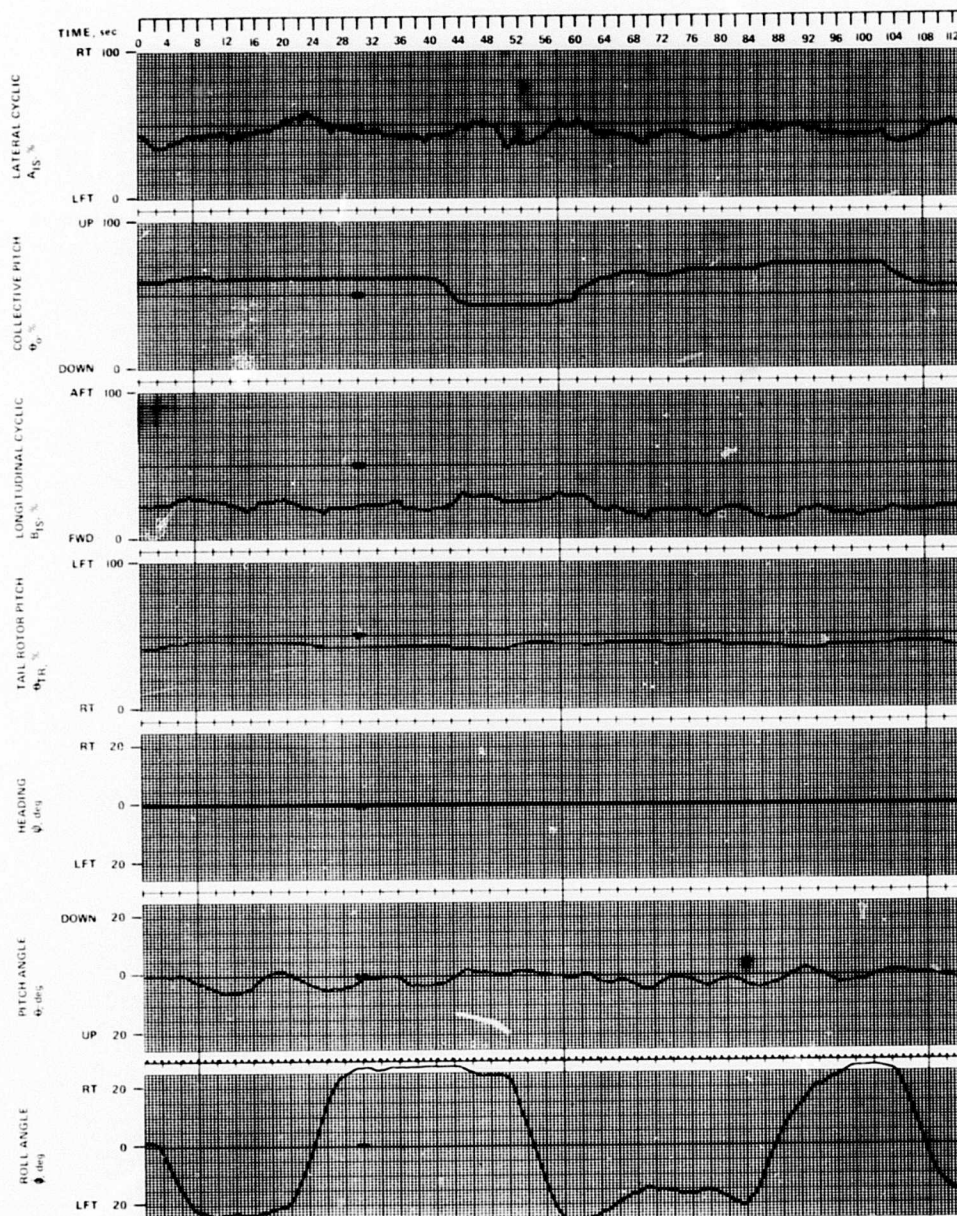
**FIGURE 82. AIRCRAFT RESPONSE TO PITCH PULSE INPUTS  
FOR GW = 47,000 LB, CG = 346 IN. AT  $V_{MAX}$ .**



**FIGURE 83. AIRCRAFT RESPONSE TO ROLL PULSE INPUTS  
FOR GW = 47,000 LB, CG = 346 IN. AT  $V_{MAX}$ .**



**FIGURE 84. AIRCRAFT RESPONSE TO YAW PULSE INPUTS  
FOR GW = 47,000 LB, CG = 346 IN. AT  $V_{MAX}$ .**



**FIGURE 85. FORWARD FLIGHT MANEUVERING - CLIMBING AND DESCENDING  
LEFT AND RIGHT TURNS FOR GW = 47,000 LB, CG = 346 IN. AT  $V_{MAX}$ .**

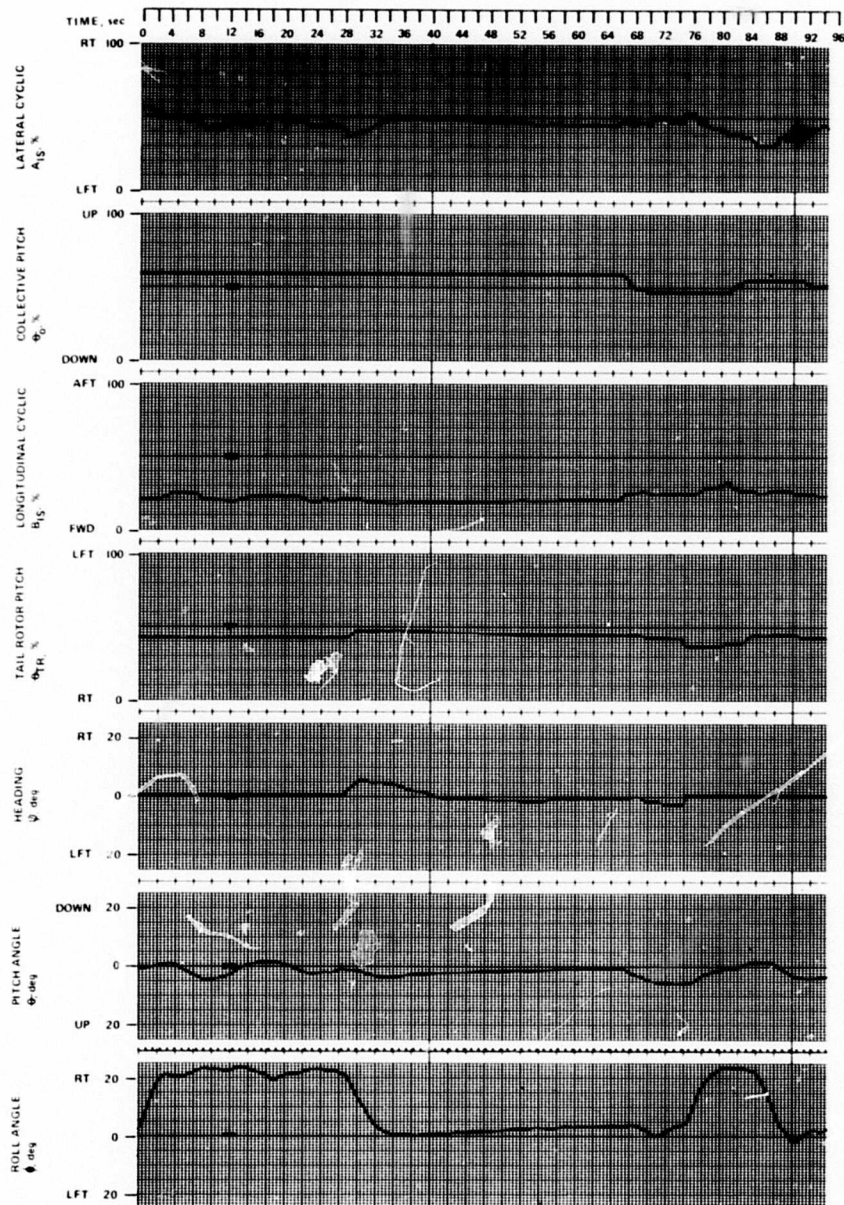
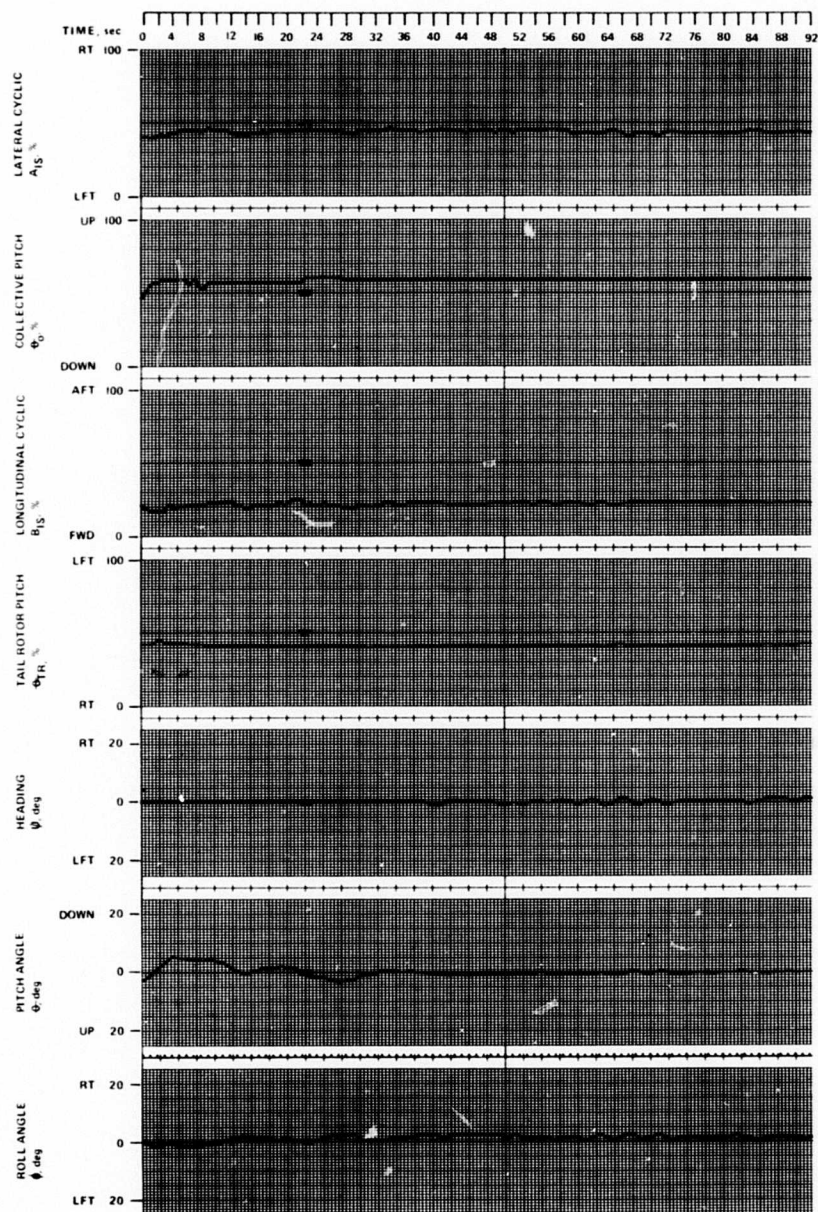


FIGURE 86. TURNS AGAINST TRIM WITH RELEASE OF ALL CONTROLS  
FOR GW = 47,000 LB, CG = 346 IN. AT  $V_{MAX}$



**FIGURE 87. STRAIGHT AND LEVEL FLIGHT AT 100 KNOTS, WIND GUST SPREAD OF 10 KNOTS FOR GW = 47,000 LB, CG = 346 IN.**

APPENDIX III

FLIGHT TEST REPORTS

No. 1

Model <u>CH-54B</u>	A/C No. <u>18462</u>	S. S. No. _____	Date <u>1-3-74</u>
Place <u>Stratford</u>		OAT/Hp <u>1</u> / <u>560</u>	
Pilots <u>Peterson</u> / _____		Winds <u>-----</u>	
Observer <u>R. Barnum</u>		C. G. <u>336</u>	
FTP <u>ETP 6411-66</u>		G. W. <u>29,000</u>	

1. CHANGES SINCE LAST REPORT: None
2. COMMENTS: This was an initial ground run for the fluidic SAS. In spite of the noise in the SAS, which could be seen at the auxiliary servo output, the run went smooth until the highest gain setting was set in. With this setting the noise in the yaw channel became noticable at 90% N<sub>R</sub>. 100% N<sub>R</sub> was reached with the other two gain settings.
3. CONCLUSIONS: Use mid-range gain valve.
4. RECOMMENDATIONS: Hover aircraft with this gain.
5. PROBLEM: Track M/R Blades

# TEST FLIGHT REPORT

No. 2

Model <u>CH-54B</u>	A/C No. <u>18462</u>	S. S. No. _____	Date <u>1-7-74</u>
Place <u>Stratford</u>	OAT/Hp <u>+6</u>	/ <u>0 to 1000</u>	
Pilots <u>J. Dixon</u>	Winds <u>-----</u>		
Observer <u>L. Cotton / R. Barnum</u>	C. G. <u>336</u>		
FTP <u>64H - 66</u>	G. W. <u>29,000</u>		

1. CHANGES SINCE LAST REPORT: None

2. COMMENTS: With gains set at 30% below nominal, pitch damping was acceptable, but there was no tendency to return to trim with SAS only, raising a question about the lagged rate system. Roll damping was low. Nominal gain resulted in still lower than normal roll damping. Gains 30% above nominal resulted in a roll instability which apparently triggered the first bending mode of the fuselage. This left us with gains slightly above nominal which resulted when the outer loop was activated in very sluggish pitch altitude retention with the selected (non-adjustable) outer loop pitch gain. Roll outer loop gain was slightly high.

Yaw damping looked acceptable, but heading hold was not obtainable.

3. CONCLUSIONS:

4. RECOMMENDATIONS:

5. PROBLEM: The oscillatory cut-off system intermittently disengaged the stick trim, yaw, and pitch/roll outer loop system. LF/ADF sense antenna standoff failed.

# TEST FLIGHT REPORT

No. 3

Model <u>CH-54B</u>	A/C No. <u>18462</u>	S. S. No. _____	Date <u>1-14-74</u>
Place <u>Stratford</u>		OAT/Hp <u>-10</u> / <u>-600</u>	
Pilots <u>Peterson</u> / <u>Bajorin</u>		Winds <u>300° 4 kts</u>	
Observer <u>L. Cotton/R. Barnum</u>		C. G. <u>29,000</u>	
FTP <u>64H-66</u>		G. W. <u>336</u>	

## 1. CHANGES SINCE LAST REPORT:

1. Helicopter was inspected for imbalance in the drive train and no cause was found.

## 2. COMMENTS: The helicopter still has a high frequency vibration at 40, 90+kts.

The fluidic SAS was evaluated in hover due to the vibration & the following was noted:

- a. SAS yaw was not compatible with the AFCS outer loop because of SAS yaw washout.
- b. SAS pitch and roll reasonable but to make pitch respond properly by increasing its gain it sustained vertical bounce which could be due to the location of the rate vortex sensor being in the cockpit rather than under the transmission.

## 3. CONCLUSIONS: Resolve basic A/C problems and evaluate SAS in forward flight.

## 4. RECOMMENDATIONS: Continue flight test.

## 5. PROBLEMS: Helicopter has a high frequency vibration in hover with #1 engine driving while in hover and with both in level flight at 40, 90+kts.

# TEST FLIGHT REPORT

		No.	<u>4</u>
Model	<u>CH-54B</u>	A/C No.	<u>18462</u>
		S. S. No.	<u>        </u>
		Date	<u>1-15-74</u>
Place	<u>Stratford</u>		
		OAT/Hp	<u>3°C / 100 ft</u>
Pilots	<u>Peterson / Craig</u>		
		Winds	<u>190° 4 knots</u>
Observer	<u>W. Fischer/R. Barnum</u>		
		C. G.	<u>356</u>
FTP	<u>64H-66</u>		
		G.W.	<u>29,000</u>

## 1. CHANGES SINCE LAST REPORT:

- A. Tail skid removed.
- B. Black M/R blade replaced.
- C. Tightened the lateral control bellcrank at the servo.

## 2. COMMENTS: The fluidic S.A.S. system with AFCS outer loop was evaluated in hover and forward flight with the following results:

Hover-Satisfactory--but heading hold was not tight enough, plus dampening in pitch was low, however, increasing it excites the first bending mode of the fuselage.

Forward Flight-Satisfactory--but roll dampening was too high and was most noticable after a right stick bump.

Landing-As a normal slow roll on touch down was being made one main gear contacted the ground first which imparted a small roll input in the aircraft. The input excited the SAS enough to sustain several oscillatory roll cycles which stopped when collective was lowered. This also indicated a deficiency in roll dampening.

NOTE: The high frequency vibration observed during the previous flights is gone. The only meaningful changes to the aircraft were the replacement of the main rotor blade, and tightened a bellcrank on the lateral control linkage.

## 3. CONCLUSIONS: Dampening settings are not optimized.

## 4. RECOMMENDATIONS: Continue basic adjustment of SAS flights.

## 5. PROBLEMS: Engagement of the SAS with the rotor turning imparts a 8" to 10" forward cyclic input. (Not a new problem).

# TEST FLIGHT REPORT

No. 5

Model <u>CH-54B</u>	A/C No. <u>18462</u>	S. S. No. _____	Date <u>1-16-74</u>
Place <u>Stratford</u>		OAT/Hp <u>4°C</u>	/ <u>190 ft</u>
Pilots <u>Peterson</u>	/ <u>Reine</u>	Winds <u>190° 8 kts.</u>	
Observer <u>Barnum</u>		C.G. <u>346"</u>	
FTP <u>64H-66</u>		G.W. <u>47,000 lbs.</u>	

1. CHANGES SINCE LAST REPORT: None

2. COMMENTS: The purpose of this flight was to quantitatively evaluate the light weight SAS dampening settings at a heavy gross weight (29,000 lbs vs 47,000 lbs). A very minor reduction in dampening was incorporated--but the response of the SAS to vertical bounce was critical in a hover. It is felt that relocation of the vortex generator sensor to the C.G. of the helicopter would improve this response. Effort should be made to eliminate the engage input (8" to 10" deflection of the tip path plane). Minneapolis Honeywell should provide a means by which individual channels can be adjusted remotely for flow and orifice size.

3. CONCLUSIONS: The dampening setting of the SAS are optimized around its installation and scope of adjustment.

4. RECOMMENDATIONS: Continue the SAS evaluation with instrumentation.

5. PROBLEMS: Incorporate a remote SAS disengage button to be held by the pilot not flying the helicopter.

# TEST FLIGHT REPORT

No. 6

Model CH-54B A/C No. 18462 S. S. No. \_\_\_\_\_ Date 1-22-74  
Place Stratford OAT/Hp 10°C / 0 ft.  
Pilots Peterson / Bajorinas Winds 300° 10 to 15 knots  
Observer R. Barnum C. G. 335 in.  
FTP 6 G. W. 28,200

1. CHANGES SINCE LAST REPORT: Yaw rate adjustment added to outer loop to make up for the lack of individual channel SAS adjustment capability.

2. COMMENTS: After outer loop gain in yaw was increased, yaw, pitch and roll channels functioned satisfactorily in hover and forward flight. However, the following were noted:

1. A step input in roll to the right of a 10° angle of bank authority would be exceeded as the helicopter returned to level flight causing a step input during recovery.
2. Lowering collective slowly from trimmed level flight of 105 and 80 kts produced an increasing nose down attitude and build in airspeed-recovery was necessary.
3. During the last portion of the flight a left pedal force of 10 lbs + was felt and would bleed off if allowed to seek a heading. This was true at 100 kts and hover.

3. CONCLUSIONS: SAS functioned satisfactory except for the above-outer loop in pitch and roll appears to need a higher rate.

4. RECOMMENDATIONS: Continue to evaluate SAS at higher G. W.'s

5. PROBLEM:

# TEST FLIGHT REPORT

No. 7

Model <u>CH-54B</u>	A/C No. <u>18462</u>	S. S. No. _____	Date _____
Place <u>Stratford</u>		OAT/Hp <u>6°C</u>	/ <u>-30 ft</u>
Pilots <u>Peterson</u>	/ <u>Bajorinas</u>	Winds _____	<u>2 kts</u>
Observer <u>Barnum</u>		C. G. <u>336 in.</u>	
FTP <u>64H-66</u>		G. W. <u>34,000 lbs</u>	

## 1. CHANGES SINCE LAST REPORT:

1. None to the SAS System
2. Water Ballast Cart attached

2. COMMENTS: The SAS system functioned satisfactorily or very much the same as it did on the previous flight on which the gross weight was 29,000 lbs. Sensitivity to vertical bounce in hover flight was increased with SAS on or off over that experience on the previous flight. Aircraft buzz and roughness also increased more than normal. The airspeed switch is intermittent. Outer loop rate still appears to be slow.

3. CONCLUSIONS: The SAS system functioned satisfactorily within its limitations of adjustment.

4. RECOMMENDATIONS: Continue SAS evaluation with high G. W. (47,000 lbs)

5. PROBLEMS: During the initial lift off to hover an excessive amount of right cyclic was required. The aircraft was weighed and found to be 300 lbs heavier on the left wheel-weight was added to the right side of the ballast cart to compensate.

# TEST FLIGHT REPORT

No. 8

Model <u>CH-54B</u>	A/C No. <u>18462</u>	S. S. No. _____	Date <u>1-23-74</u>
Place <u>Stratford</u>		OAT/Hp <u>10°C</u>	/ <u>+125 ft</u>
Pilots <u>Peterson /</u>	<u>Bajorinas</u>	Winds <u>15 kts</u>	<u>180°</u>
Observer <u>R. Barnum</u>		C.G. <u>346 in</u>	
FTP <u>64H-66</u>		G.W. <u>47,000 lbs.</u>	

1. CHANGES SINCE LAST REPORT: None

2. COMMENTS: Very little change was observed in the performance of the SAS at this gross weight as compared to the previous flights of 35,000 and 29,000 lbs. Rolling into a turn (20° AOB) against stick trim and holding it for 90° of heading change then releasing both cyclic and pedal switches simultaneously resulted in a recovery to level flight which was comparable to the experienced with the electronic AFCS.

In summary, because this was the last flight of this evaluation program, it is concluded that the system does perform satisfactorily, however, due to the lack of individual channel adjustment and only remote adjustment of yaw in the outer loop. It is felt that the SAS + outer loop combination of rate/gain could be optimized if these adjustable parameters were made available.

Turning the fluidics on, with the aircraft static, produced small control motions in the output rods of the auxiliary servo. Although this motion was not perceptible to the pilot it was, nevertheless, felt to be objectionable.

The engagement transients of the SAS are unacceptable.

The fluidic system was audible over aircraft noise with a helmet on.

3. CONCLUSIONS: The fluidic SAS functioned satisfactorily.

4. RECOMMENDATIONS: Follow thru with summary comments.

5. PROBLEMS: As mentioned in summary - none with the aircraft.

# LIST OF SYMBOLS

$\theta_f$	Pitch attitude, deg
$\dot{\theta}_f$	Pitch attitude rate, deg/sec
$\phi$	Roll attitude, deg
$\dot{\phi}$	Roll attitude rate, deg/sec
$\psi$	Heading, deg
$\dot{\psi}$	Heading rate, deg/sec
$B_{IS}$	Longitudinal cyclic, deg (or percent)
$A_{IS}$	Lateral cyclic, deg (or percent)
$\theta_o$	Collective pitch, deg (or percent)
$\theta_{TR}$	Tail rotor collective pitch, deg (or percent)
$S$	La Place operator
$I_X$	Roll inertia
$I_Y$	Pitch inertia
$I_Z$	Yaw inertia